

DYNAMIC STABILITY TEST RESULTS

ON AN 0.024 SCALE B-1 AIR VEHICLE

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DYNAMIC STABILITY TEST RESULTS
ON AN 0.024 SCALE B-1 AIR VEHICLE

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FOREWORD

This report was prepared by the Aerodynamics Group of the Los Angeles Division of North American Rockwell Corporation at Los Angeles, California under Air Force contract No. F33657-70-C-0800, and the NASA Vehicle Dynamics Section of Langley Research Center, Langley Air Force Base, Virginia.

Three Langley Research Center wind tunnels were utilized to obtain the main damping derivatives (C_{mq} , C_{nr} and C_{lp}) over the Mach number range of the air vehicle - the Langley 4 foot Unitary Plan wind tunnel (supersonic), the Langley 8 foot Transonic Pressure tunnel and the Langley 7 x 10 foot High Speed tunnel (transonic and subsonic).

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NASA, LRC, Vehicle Dynamics Section personnel conducting the test were Messrs. R. A. Kilgore, E. E. Davenport, J. Adcock and R. Boyden under the supervision of Mr. H. G. Wiley.

SUMMARY

Dynamic longitudinal and lateral-directional stability characteristics of the B-1 air vehicle have been investigated in three wind tunnels at the Langley Research Center. The main rotary derivatives were obtained for an angle of attack range of -3 degrees to +16 degrees for a Mach number range of 0.2 to 2.16. Damping in roll data could not be obtained at the supersonic Mach numbers. The Langley 7 x 10 foot high speed tunnel, the 8 foot transonic pressure tunnel and the 4 foot Unitary Plan wind tunnel were the test sites. An 0.024 scale light-weight model was used on a forced oscillation type balance. Test Reynolds number varied from $0.474 \times 10^6/\text{Ft.}$ to $1.55 \times 10^6/\text{Ft.}$ through the Mach number range tested.

The results of the investigation showed that the dynamic stability characteristics of the model in pitch and roll were generally satisfactory up to an angle attack of about +6 degrees.. In the wing sweep range from 15 to 25 degrees the positive damping levels in roll deteriorated rapidly above +2 degrees angle of attack. This reduction in roll damping is believed to be due to the onset of separation over the wing as stall is approached.

In the subsonic and transonic speed range, yaw damping levels are negative in some cases (wing $\Lambda = 25^\circ, 55^\circ$ and 65°), vertical tail inputs are zero to negative and body input is about twice the estimate. The afterbody of the rotary derivative model is distorted from the air vehicle lines to permit insertion of the balance and to provide clearance between the base and the sting for model oscillation.

The resulting large diameter base is believed to have caused an aerodynamic interference between the model afterbody and the horizontal and vertical tails which significantly altered the vertical tail input to yaw damping. Since the B-1 fuselage has a much different shape than the model in the vicinity of the vertical tail, the yaw damping characteristics measured on the model are probably not representative of the B-1 air vehicle.

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NOMENCLATURE

Symbols

l	reference length = 10.33 ft.
C_l	rolling moment coefficient, $\frac{\text{Rolling moment}}{q_\infty S l}$
$C_{lP} = \frac{\partial C_l}{\partial \left(\frac{P l}{2V}\right)}$	per radian
C_m	pitching moment coefficient, $\frac{\text{Pitching moment}}{q_\infty S l}$
k	reduced frequency parameter, $\frac{\omega l}{2V}$ in pitch and yaw
M	freestream Mach number
q	angular velocity of model about body Y-axis, rad/sec
q_∞	freestream dynamic pressure, psf
r	angular velocity of model about body Z-axis, rad/sec
S	reference area = 60.44 ft. ²
V	freestream velocity ft/sec
α	angle of attack, degrees or radians; mean angle of attack, deg.
β	angle of sideslip, degrees or radians; mean angle of sideslip, degrees
ω	angular velocity, $2\pi f$, rad/sec
$C_{mq} = \frac{\partial C_m}{\partial \left(\frac{q l}{2V}\right)}$	/rad
$C_{mq} + C_{m\dot{\alpha}}$	damping in pitch parameter, per radian

$$C_{m\dot{q}} = \frac{\partial C_m}{\partial \left(\frac{\dot{q} \ell^2}{4V^2} \right)} / \text{rad}$$

$$C_{m_\alpha} = \frac{\partial C_m}{\partial \alpha} / \text{rad}$$

$$C_{m_\alpha} - k^2 C_{m\dot{q}} \quad \text{oscillatory-longitudinal-stability parameter, per radian}$$

$$C_{m\dot{\alpha}} = \frac{\partial C_m}{\partial \left(\frac{\dot{\alpha} \ell}{2V} \right)} / \text{rad}$$

$$C_n \quad \text{yawing-moment coefficient, } \frac{\text{Yawing moment}}{q_\infty S \ell}$$

$$C_{nr} = \frac{\partial C_n}{\partial \left(\frac{r \ell}{2V} \right)} / \text{rad}$$

$$C_{nr} - C_{n\dot{\beta}} \cos \alpha \quad \text{damping-in-yaw parameter, per radian}$$

$$C_{n\dot{r}} = \frac{\partial C_n}{\partial \left(\frac{\dot{r} \ell^2}{4V^2} \right)} / \text{rad}$$

$$C_{n_\beta} = \frac{\partial C_n}{\partial \beta} / \text{rad}$$

$$C_{n_\beta} \cos \alpha + k^2 C_{n\dot{\beta}} \quad \text{oscillatory-directional-stability parameter, per radian}$$

$$C_{n\dot{\beta}} = \frac{\partial C_n}{\partial \left(\frac{\dot{\beta} \ell}{2V} \right)} / \text{rad}$$

$$C_{lp} + C_{l\dot{\beta}} \sin \alpha \quad \text{damping in roll parameter, per radian}$$

$$C_{l\dot{\beta}} = \frac{\partial C_l}{\partial \left(\frac{\dot{\beta} b}{2V} \right)}$$

INTRODUCTION

Design of guidance and control systems for aerodynamic configurations requires knowledge of the various dynamic stability derivatives through wide ranges of flight speeds. Experimental data and theories are available at low speeds, and a limited amount of data is available at supersonic speeds. However, at transonic speeds no adequate theories for predicting these derivatives are available, especially at angle of attack, and little experimental data exists.

Therefore, wind tunnel test investigations have been conducted by the National Aeronautics and Space Administration and North American Rockwell Corporation to substantiate the levels of the main dynamic stability derivatives of the B-1 air vehicle. The information obtained will be used to determine their effect on the flight dynamics of the air vehicle.

DISCUSSION

MODEL DESCRIPTION

The model is a relatively lightweight 0.024-scale replica of the B-1 air vehicle with provision for sting mounting to the NASA-Langley rigidly forced oscillation system. Aerodynamic similarity is maintained except for the fuselage modifications necessary to accommodate the support sting and oscillating mechanism, figures 1 through 8. The upper fuselage moldline follows the air vehicle lines, reference 1, but a significant deformation of the lower surface was necessary to form the required circular cross section aft of fuselage station 950. Forward of this station, the circular section gradually blends into the air vehicle mold lines at fuselage station 150.

The wing and empennage surfaces are fabricated of aluminum alloy and the forward fuselage section is constructed of laminated fiberglass. The wing is pivoted to provide four discrete sweep positions and the horizontal tail is adjustable to -10° , 0° , and $+10^\circ$ deflection.

The balance mounting bulkhead can be moved in the mounting tube to place the balance center-of-rotation at the longitudinal position corresponding to the full scale air vehicle center of gravity for the various wing sweep positions.

The reference dimensions used in the data calculation are the following:

<u>0.24-Scale</u>	<u>Full Scale</u>
$S_w = 1.1209$ sq. ft.	1946 sq. ft.
$b_w = 39.364$ in.	1640.17 in.
$\bar{c}_w = 4.417$ in.	184.05 in.
F.S. of $.25\bar{c}_w = 23.708$ in.	987.85 in.
F.S. of $.45\bar{c}_w = 24.59$ in.	1024.66 in.
F.S. of $.75\bar{c}_w = 25.92$ in.	1079.88 in.

Additional model and test information is contained in references 2 and 3.

WIND TUNNELS

Three wind tunnels were used to obtain the data presented herein. All three are equipped for control of relative humidity and total temperature of the air in the tunnel to minimize the effects of condensation and for control of total pressure in order to obtain the test Reynolds number.

Langley Unitary Plan Wind Tunnel.-The data at Mach numbers 1.8 and 2.16 were obtained in test section number 1 of the Langley Unitary Plan wind tunnel. This single return tunnel has a test section about 4.0 feet by 4 feet. An asymmetric sliding block, which varies the area ratio, is used to change Mach number from about 1.47 to 2.86. The angle of attack mechanism used for this investigation has a total range of about 25 degrees when used with the oscillation balance mechanism.

Langley 8-foot Transonic Pressure Tunnel.-The data at Mach numbers from 0.40 to 1.20 were obtained in the Langley 8-foot transonic pressure tunnel. The test section of this single return closed-circuit wind tunnel is about 7.0 feet by 7 feet with slotted upper and lower walls to permit continuous operation through the transonic speed range. Test section Mach numbers from about 0.2 to 1.30 can be obtained and kept constant by controlling the speed of the tunnel fan drive motor. The Mach number is reasonably uniform throughout the test section, with a maximum deviation from the average freestream Mach number of approximately 0.01 at the higher Mach numbers.

The sting support strut is designed to keep the model near the center line of the tunnel through a range of sting angles of attack from about -4 to +22 degrees when used with the oscillation balance mechanism.

Langley 7 x 10 foot High Speed Tunnel.-Correlation data for Mach numbers from 0.2 to 0.8 were obtained in the Langley 7 by 10 foot high speed tunnel. It is a single return wind tunnel which operates with atmospheric stagnation pressure and has a conventional, closed, rectangular fixed geometry test section. The sting support strut is designed to keep the model near the center line of the tunnel through a range of sting angles of attack from about -12 to +16 degrees when used with the oscillation balance mechanism.

OSCILLATION BALANCE MECHANISM

Two oscillation balance mechanisms were used for these tests, one for pitch and yaw and the other for roll. The damping terms are measured as a function of how much torque is required to oscillate the models at a certain amplitude. A description of these mechanisms is given in reference 3.

AXIS SYSTEM

The aerodynamic parameters are referred to the body system of axes originating at the oscillation center of the model as shown in figure 7. The reference dimensions are based on the geometric characteristics of the model with the wings in the swept forward position ($\Lambda = 15^\circ$) regardless of the actual test wing-sweep position.

PRESENTATION OF RESULTS

The actual run schedules for the five tests presented in this report are shown on Tables I through V (pages 20 through 25). The tests were as follows:

<u>Table</u>	<u>Test No.</u>	<u>Tunnel</u>	<u>Derivative</u>
I	952	4' UPWT	Pitch & Yaw
II & III	596	8' TPT	Pitch & Yaw
IV	599	8' TPT	Roll
V	934	7 x 10' HST	Yaw & Pitch

Test Reynolds number, based on the wing mean aerodynamic chord was approximately 1,100,000 in the Unitary Plan tunnel, 1,550,000 in the 8' TPT, and between 1,100,000 and 1,450,000 in the 7 x 10 tunnel with the exception of the data taken at Mach 0.2, which was at a Reynolds number of 500,000.

The damping in pitch derivative $C_{m\dot{\alpha}} + C_{m\ddot{\alpha}}$ and the oscillatory-stability parameter $C_{m\alpha} - k^2 C_{m\dot{\alpha}}$ are presented in figures 9 through 53 for subsonic, transonic and supersonic Mach numbers. The data obtained show positive damping and are in fair agreement with estimated levels at zero angle of attack, and in general show an increase in damping with increase in angle of attack. Component build-up data, such as figures 13 thru 19, show that the horizontal tail contributes approximately 50 percent of the damping in pitch. Plots showing the effect of wing leading edge sweep, Mach number, center of gravity shift and wing input on the damping in pitch parameters are also presented. The effects of Mach number at constant angles of attack are presented on figures 9 through 12 for various leading edge sweep angles. Estimated data are superimposed on these plots.

The damping in yaw derivative $C_{n\dot{\beta}} - C_{n\ddot{\beta}}$ and the oscillatory stability parameter $C_{n\beta} \cos + k^2 C_{n\dot{\beta}}$ are presented on figures 54 through 102 for subsonic, transonic and supersonic Mach numbers. Inspection of the supersonic data reveal that $C_{n\dot{\beta}} - C_{n\ddot{\beta}}$ measured approximately 50 percent more damping than was estimated. The increased damping was due primarily to body input, which was probably influenced by the enlarged model base. Vertical tail input to the damping in yaw parameter was about 70 percent of the estimated data. This reduction in tail damping may be due in part to the distorted model afterbody, as suspected in the subsonic tests. Directional stability, $C_{n\beta}$ due to the tail was slightly less than that measured in the transonic wind tunnel.

The subsonic damping in yaw data measured in the Langley 8' TPT was questionable because of variation with frequency, and zero or negative damping due to vertical tail. The model was shifted to the Langley 7 x 10 HST for further investigation with a stiffer model support system. The variation of $C_{n_r} - C_{n_{\dot{\beta}}}$ with frequency was reduced but the vertical tail input to yaw damping was still zero or negative. Removing the horizontal tail improved the vertical tail damping input. A fairing was tested on the vertical tail in the area between the horizontal tail and aft fuselage mold line, page 32, to partially simulate the three dimensional relief between the horizontal tail and fuselage as it exists on the air vehicle. Data obtained with this "channel filler block" brought the tail on yaw damping level closer to estimated values, fig 68 & 69. Tail input, however, was still less than 50 percent of estimated. The data also show that damping in yaw gets less positive with increase in Mach number from 0.2 to 0.8 for all sweeps. Installation of the "filler blocks" reversed this trend through this Mach number range. Since either the removal or deflection of the horizontal tail, or the addition of the filler blocks on the vertical tail drastically altered the vertical tail input to yaw damping, it is apparent that aerodynamic interference between the fuselage afterbody, horizontal and vertical tails significantly affected the yaw damping of the model configuration. Since the model lines are significantly different than the B-1 air vehicle in this area, the yaw damping characteristics measured on the model are probably not representative of the B-1 air vehicle.

The damping in roll derivative $C_{l_p} + C_{l_{\dot{\beta}}} \sin \alpha$ is presented on figures 103 through 119. These data were obtained at subsonic through transonic Mach numbers for the wing leading sweep range of the air vehicle. Damping data obtained with the wing in the aft sweep positions, $\Lambda = 65$ and 55 degrees show good agreement with estimated data at a fairly constant level of damping with variation of angle of attack. Data obtained with the forward leading edge sweep positions $\Lambda_{LE} = 25$ and 15 degrees show good agreement with estimated data from -3 to 0 to 2 degrees angle of attack. The damping levels drop rapidly with increasing angle of attack above 0 to 2 degrees. This phenomenon is associated with the onset of airflow separation over the wing as the angle of attack for stall is approached. Force data obtained on other B-1 models indicate the onset of separation at somewhat higher angles of attack than those at which the roll damping begins to deteriorate. This is probably because the force models were tested at higher Reynolds numbers than the rotary derivative model.

CONCLUSIONS

Data obtained with the wing panels swept 15 and 25 degrees showed positive damping in pitch, that generally remained linear with angle of attack for angles below about six degrees. The damping in roll data obtained showed positive damping at angles of attack from -2 to +2 degrees; above these angles of attack the damping deteriorated rapidly. This phenomenon is closely associated with the onset of separation at these sweeps and angles of attack.

Data obtained with the wing panels swept 55 and 65 degrees showed positive damping in pitch and roll and remained linear with angle of attack for angles below about six degrees at subsonic and transonic Mach numbers and about 10 degrees supersonically. No supersonic damping in roll data was obtained.

The aerodynamic damping in pitch and roll measured in these tests appear reasonable, and are substantiated by estimates for the B-1 configuration.

The measured aerodynamic damping in yaw showed some unusual results. The damping due to the body was larger than estimated, but the measured damping due to the vertical tail was less than estimated, and sometimes even negative at Mach numbers between .7 and .8. Component buildup showed the vertical tail input to be strongly affected by aerodynamic interference between the fuselage afterbody and the horizontal and vertical tails. Since the model shape is significantly different than the B-1 air vehicle in this area, the yaw damping characteristics measured on the model are probably not representative of the B-1 air vehicle.

RECOMMENDATIONS

In order to obtain data that can be used during design to define the flight dynamics of the B-1 air vehicle, additional testing must be accomplished. The model lines should be modified to more closely approximate B-1 afterbody lines so that the questionable validity of the initial test data can be clarified. Since the horizontal tail planform was changed subsequent to the construction of this model, this component should also be updated. Also since roll damping was not attainable above $M = 1.2$, additional testing for this parameter should be accomplished over the complete Mach range.

DIMENSIONAL DATA

W ₄₇	Wing	
	Model	
	Reference Area, Ft. ²	1.120
	Span, Ft.	3.272
	Aspect Ratio	9.560
	Taper Ratio	.351
	Chords, In.:	
	Root (B.P. 0.0)	6.075
	Tip (B.P. 19.634)	2.135
	M.A.C. (B.P. 8.247)	4.420
	Fus. Sta. of 0.25 M.A.C., In.	23.705
	Fus. Sta. of Wing Pivot (B.P. 3.480), In.	23.256

Leading Edge Sweepback Angle, Deg.	14.992°
Dihedral, Deg.	-1.940°
Incidence, Deg.	0.0°

h ₅₈	Wing hood	
	Data for One of Two Sides	
	Model	
	Area, Ft ²	.065
	Corresponding Wing Area, ft ²	1.120
	Wing Sweep, Deg.	15.

B ₅₁	Body	
	Model	
	Length, Ft.	3.40
	Max. Width (F.S. 22.8-40.8), In.	3.6
	Max. Depth (F.S. 22.8-40.8), In.	3.6
	Max. Cross-Sectional Area (F.S. 22.8-40.8), Ft ²	.071
	Fineness Ratio	11.333

B₆₀ B₅₁ with Ogive Forebody

B₆₁ B₅₁ with Filler Blocks in Channel Between Horizontal Tail and Fuselage (see page 32)

B₆₂ B₅₁ with Vertical Tail Mounting Plate Faired Forward to Blend More Smoothly into Fuselage

TR 19 Boundary Layer Transition Grit
No. 100 Carborundum Grit
Wing, Horizontal, Vertical (Both Sides)
Width = .05 \pm In.
.5 In Aft of L.E.
Fuselage Forebody
Width = .05 In
1.5 in Aft. of L.E.

N54 Nacelle
Data for 1 of 2 Sides

Model	
Length (Overall), Ft.	.895
Max. Width, (F.S. 29.39), In.	3.03
Max. Depth, (F.S. 28.20), In.	1.65
Inlet Area, Ft. ²	.0124
Capture Area, Ft. ²	.0124
Exit Area (Total), Ft. ²	.00863
Fus. Sta. of Nacelle Leading Edge, In.	21.77
Nacelle Centerline	
B.P., In. (F.S. 21.77), In.	3.386
W.P., In. (F.S. All), In.	-.840
Offset Angle Nac. CL (L.E. Inb'd)	.318°

C₁₂ Structural Model Control Vane

Total

Area, Ft ²	.027
Span (Equiv.), Ft.	.25
Aspect Ratio	2.298
Taper Ratio	.100
Chords, Inc.	
Root (B.P. 0.0)	2.053
Tip (B.P. 1.499)	.206
M.A.C. (B.P. .545)	1.381
Fus. Sta. of 0.25 M.A.C. (W.P. .417), In.	5.273
Dihedral Angle, Deg.	-30.0
Incidence Angle, Deg.	0.
Sweepback Angle, Deg.	
Leading Edge	35.022°
0.25 Chord Element	23.458°
Airfoil Section L.E.R. = (.051) (Local Chord)	
T.E.R. = Knife Edge	

Exposed

Area, Ft ²	.0065
Span (Equiv.), Ft.	.110
Aspect Ratio	1.867
Taper Ratio	.202
Chords, In.	
Root (B.P. .837)	1.021
Tip (B.P. 1.499)	.206
MA.C. (B.P. 1.095)	.704
Fus. Sta of 0.25 M.A. C. (W.P. .0999), In.	5.548

H₁₆₃ Horizontal Tail

Total

Area, Ft. ²	.293
Span (Equiv.), Ft.	1.071
Aspect Ratio	3.918
Taper Ratio	.304
Chords, In.	
Root (B.P. 0.0)	5.030
Tip (B.P. 1.528)	1.528
M.A.C. (B.P. 2.640)	3.591
Fus. Sta. of 0.25 M.A.C. (W.P. 3.024), In.	38.395
Angles, Deg.	
Dihedral	0.0
Incidence	0.0
Leading Edge Sweep	32.500
0.25 Chord Element	26.601
Airfoil Section	
Root (B.P. 0.0)	65A007
Tip (B.P. 6.457)	65A007

Exposed

Area, Ft. ²	.271
Span (Equiv.), Ft.	1.017
Aspect Ratio	3.824
Taper Ratio	.315
Chords, In.	
Root (B.P. .321)	4.855
Tip (B.P. 6.42)	1.528
M.A.C. (B.P. 2.842)	3.481
Fus. Sta. of 0.25 M.A.C. (W.P. 3.024), In.	38.497

V₄₆ Vertical Tail

Area, Ft. ²	.146
Span (Equiv.), Ft.	.409
Aspect Ratio	1.184
Taper Ratio	.304
Equivalent Chords, In.	
Root (W.P. 2.016)	6.362
Tip (W.P. 6.930)	1.937
M.A.C. (W.P. 4.036)	4.543
Fus. Sta. of 0.25 M.A.C. (B.P. 0.0), In.	34.049
Angles, Deg.	
Cant	0.0
Offset	0.0
Leading Edge Sweep	50.777
0.25 Chord Element	45.0
Airfoil Section	
Root (W.P. 2.016)	65A010
(W.P. 3.854)	65A010
Tip (W.P. 6.962)	65A005

D₃ Dorsal Fin

Area, Ft. ²	.021
Length, In.	11.562

REFERENCES

1. D481-55B-1, " General Arrangement, Three View Drawing of the B-1 Air Vehicle" Dated 14 December 1970.
2. NA-70-550-2, TP/PS 0221-1-002, "Pretest Information for B-1 0.024-Scale Air Vehicle Rotary Derivative Tests in Langley 4 Foot UPWT", Dated 19 April 1971.
3. NA-70-550-2, TP/PS 0221-1-001, "Pretest Information for B-1 0.024-Scale Air Vehicle Rotary Derivative Tests in Langley 8-foot TPT", Dated 17 May 1971.
4. NASA TN D-1231, "A Rigidly Forced Oscillation System for Measuring Dynamic-Stability Parameters in Transonic and Supersonic Wind Tunnels", Dated March 1962.

TABLE I
LANGLEY 4' UPWT TEST NO. 952
ACTUAL RUN SCHEDULES - PITCH AND YAW OSCILLATIONS

CONFIGURATION	Λ	δ_H	C.G. POS. %cw	PITCH MACH NO./RUN NO.		YAW MACH NO./RUN NO.	
				1.8	2.16	1.8	2.16
W47h58B51N54C12D3H163V46 TR19	65°	0°	.27	2	1	22	21
W47h58B51N54C12D3H163V46 TR19	65°	0°	.75	4	3	24	23
W47h58B51N54C12D3--V46 TR19	65°	-	.75	7	6	-	-
W47h58B51N54C12--- TR19	65°	-	.75	-	-	27	26
W47h58B51N54C12D3H163V46 TR19	65°	-10°	.75	9	8	29	28
W47h58B51N54C12D3H163V46 TR19	55°	0°	.75	11	10	31	30
W47h58B51N54C12D3--V46 TR19	55°	-	.75	13	12	-	-
W47h58B51N54C12----- TR19	55°	-	.75	-	-	33	32
B51C12 TR19	-	-	.75	16	14	-	-
B51C12D3H163V46 TR19	-	0°	.75	18	17	20	19

TABLE II
LANGLEY 8' TPT TEST NO. 596
ACTUAL RUN SCHEDULE - PITCH OSCILLATION

CONFIGURATION	Λ	δ_H	C.G. POS. %CW	MACH NUMBER/RUN NUMBER						
				.55	.7	.75	.85	.95	1.03	1.2
W47h58B51N54C12 D3H163V46 TR19	55°	0°	.75	3	-	-	2	5	4	1
W47h58B51N54C12 D3H163V46 TR19	65°	0°	.75	6	-	-	7	10	9	8
W47h58B51N54C12D3--V46 TR19	65°	-	.75	15	-	-	14	13	12	11
W47h58B51N54C12D3H163V46 TR19	65°	0°	.25	20	-	-	19	18	17	16
W47h58B51N54C12D3H163V46 TR19	15°	0°	.25	23	22	21	-	-	-	-
W47h58B51N54C12D3--V46 TR19	15°	-	.25	26	25	24	-	-	-	-
W47h58B51N54C12D3--V46 TR19	25°	-	.45	29	28	27	-	-	-	-
W47h58B51C12D3H163V46 TR19	25°	0°	.45	32	31	30	-	-	-	-
B51C12D3H163V46 TR19	-	0°	.75	37	-	-	36	35	34	33
B51C12D3---V46 TR19	-	-	.75	39	-	-	38	-	-	-

TABLE III
LANGLEY 8' TPT TEST NO. 596 (CONT'D)
ACTUAL RUN SCHEDULE - YAW OSCILLATION

CONFIGURATION	A	δ_H	C.G. POS. % \bar{c}_w	MACH NUMBER/RUN NUMBER						
				.55	.7	.75	.85	.95	1.03	1.2
W47h58B51N54C12D3H163V46 TR19	55°	0	.75	44	—	—	43	42	41	40
W47h58B51N54C12D3H163V46 TR19	65°	0	.75	49	—	—	48/92**	47	46	45
W47h58B51N54C12----- TR19	65°	—	.75	54	—	—	53	52	51	50
W47h58B51N54C12D3H163V46 TR19	65°	0°	.25	60	—	59	58	57	56	55
W47h58B51N54C12D3H163V46 TR19	15°	0°	.25	63	62	61	—	—	—	—
W47h58B51N54C12----- TR19	15°	—	.25	66	65	64	—	—	—	—
W47h58B51N54C12----- TR19	25°	—	.45	69	68	67/89*	—	—	—	—
W47h58B51N54C12D3H163V46 TR19	25°	0°	.45	72/91**	71	70/88*	—	—	—	—
W47h58B51---C12D3H163V46 TR19	65°	0°	.75	76	—	75	74	—	—	73
B51C12D3H163V46 TR19	—	0°	.75	81	—	—	80	79	78	77
B51C12 TR19	—	—	.75	86	—	—	85	84	83	82
W47h58B51N54C12D3H163V46 TR19	25°	0°	.45	—	88*	87*	—	—	—	—

*Runs 87, 88 and 89 Made with Pitch Oscillation Rig with Model at $\phi = 270^\circ$

**Runs 90, 91 and 92 Made with 2 more Cables added to the Yaw Oscillation Rig to increase System Stiffness

TABLE IV
LANGLEY 8' TPT TEST NO. 599
ACTUAL RUN SCHEDULE - ROLL OSCILLATION

CONFIGURATION	Λ	δ_H	C.G. Pos %cw	MACH NUMBER/RUN NUMBER						
				.55	.7	.75	.85	.95	1.03	1.2
W47h58B51N54C12D3H163V46 TR19	65°	0°	.75	5	—	—	4	1	3	2
W47h58B51N54C12D3--V46 TR19	65°		.75	10	—	—	9	8	7	6
W47h58B51N54C12D3H163V46 TR19	55°	0°	.75	15	—	—	13	12	14	11
W47h58B51N54C12D3H163V46 TR19	65°	0°	.25	20	—	—	18	17	19	16
W47h58B51N54C12D3H163V46 TR19	15°	0°	.25	23	22	21	—	—	—	—
W47h58B51N54C12D3--V46 TR19	15°	—	.25	26	25	24	—	—	—	—
W47h58B51N54C12D3--V46 TR19	25°	—	.45	29	28	27	—	—	—	—
W47h58B51C12D3H163V46 TR19	25°	0°	.45	32	31	30	—	—	—	—
W47h58B51N54C12D3H163V46 TR19	25°	-10°	.45	35	34	33	—	—	—	—

TABLE V
LANGLEY 7 x 10' HST TEST NO. 934
ACTUAL RUN SCHEDULE - YAW OSCILLATION

CONFIGURATION		Λ	δ_H	C.G.POS. % \bar{c}_w	MACH NUMBER/RUN NUMBER				
					.20	.55	.7	.75	.8
W47h58B51N54C12D3H163V46	TR19	25°	0°	.45	—	6/12,*68	5/69*	4	—
W47h58B51N54C12-----	TR19	25°	—	.45	—	9	8	7	—
W47h58B51N54-----	TR19	25°	—	.45	—	—	10	—	—
W47h58B51N54--D3H163V46	TR19	25°	0°	.45	—	—	11	—	—
W47h58B51N54C12D3H163V46	TR19	15°	0°	.25	—	15	14	13	—
W47h58B51N54C12-----	TR19	15°	—	.25	—	18	17	16	—
B51C12D3H163V46	TR19	—	0°	.75	—	22	21	20	19
B51C12D3--V46	TR19	—	—	.75	—	23	24	25	26
B51C12	TR19	—	—	.75	—	27	—	28	—
B60	TR19	—	—	.75	—	30	—	29	—
B60D3--V46	TR19	—	—	.75	—	31	—	32	—
B60D3H163V46	TR19	—	0°	.75	—	33	—	34	—
B51D3H163V46	TR19	—	0°	.75	—	36	—	35	—
W47h58B51N54C12D3H163V46	TR19	65°	0°	.75	41	37	38	39/67*	40
W47h58B51N54C12D3H163V46	TR19	65°	-10°	.75	—	42	—	43	—
W47h58B51N54C12-----	TR19	65°	—	.75	—	44	—	—	45
W47h58B51N54C12D3--V46	TR19	65°	—	.75	—	46	—	48	47
W47h58B51N54C12D3H163V46	TR19	55°	0°	.75	—	49/**	52	51/**	50/**

TABLE V (CONT'D)

CONFIGURATION		Λ	δ_H	C.G.POS. % \bar{c}_w	MACH NUMBER/RUN NUMBER				
					.20	.55	.70	.75	.80
W47h58B51N54C12D3H163V46	TR19	65°	0°	.75	—	56/65*	63	57/64*	62
W47h58B61N54C12D3--V46	TR19	65°	—	.75	—	58	—	59	—
W47h58B51N54C12D3H163V46	TR19	65°	+10°	.75	—	61	—	60	—
W47h58B62N54C12D3H163V46	TR19	65°	0°	.75	—	—	—	66	—
W47h58B61N54C12D3H163V46	TR19	25°	0°	.45	—	72	73	—	—
W47h58B61N54C12D3--V46	TR19	25°	—	.45	—	74	75	—	—
W47h58B51N54C12D3--V46	TR19	25°	—	.45	—	71	70	—	—

*Runs 12, 64, 65, and 68 are Check Runs

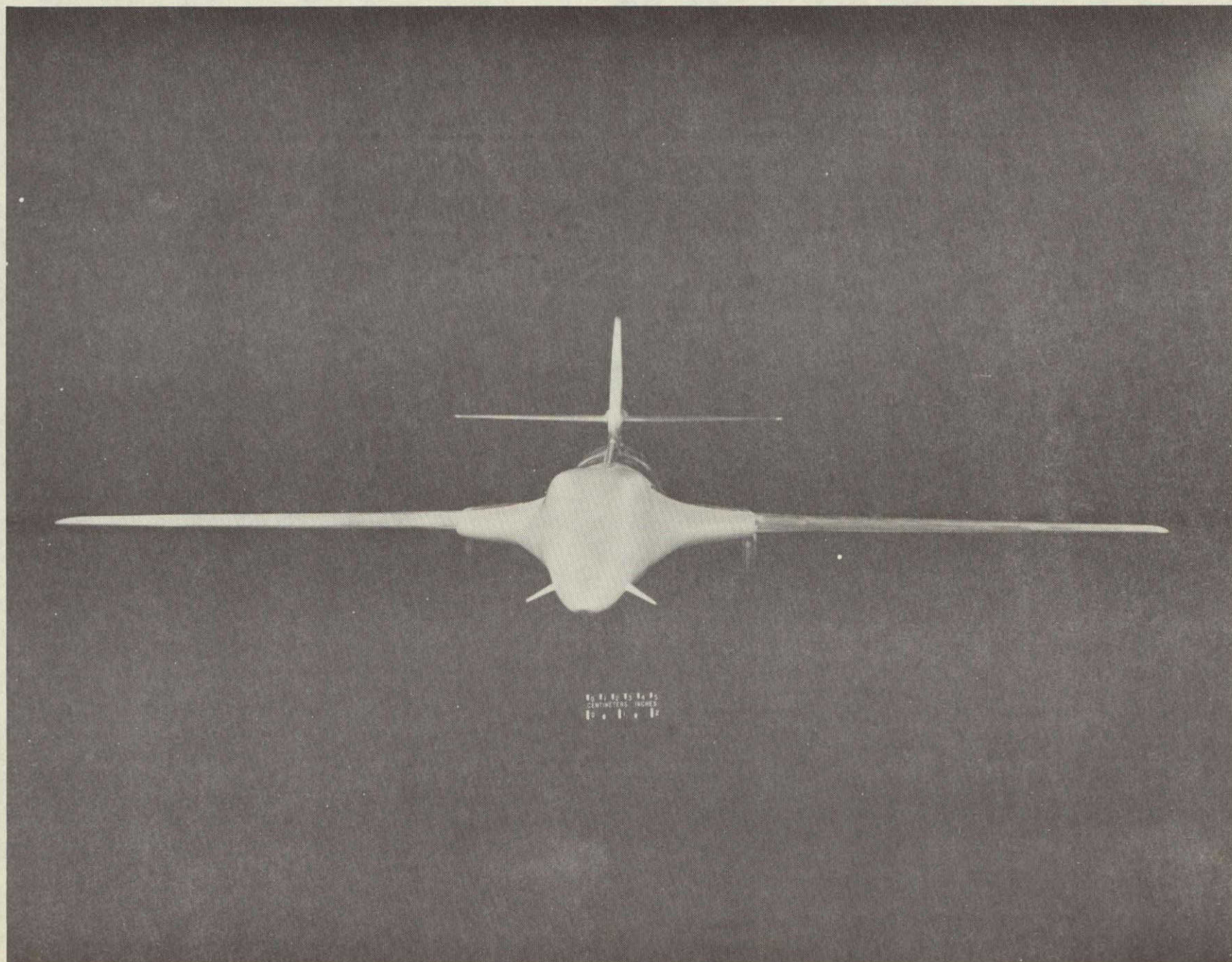
**Runs 53, 54 and 55 were made at $\Lambda = 4^\circ$ only with Horizontal Steel Rod Stiffness Instead of Cable

LANGLEY 7 x 10' HST TEST NO. 936

ACTUAL RUN SCHEDULE - PITCH OSCILLATION

CONFIGURATION		Λ	δ_H	C.G.POS. % \bar{c}_w	MACH NUMBER/RUN NUMBER				
					.20	.55	.70	.75	.80
W47h58B51N54C12D3H163V46	TR19	25°	0°	.45	—	1	2	—	—
W47h58B61N54C12D3H163V46	TR19	25°	0°	.45	—	3	4	—	—

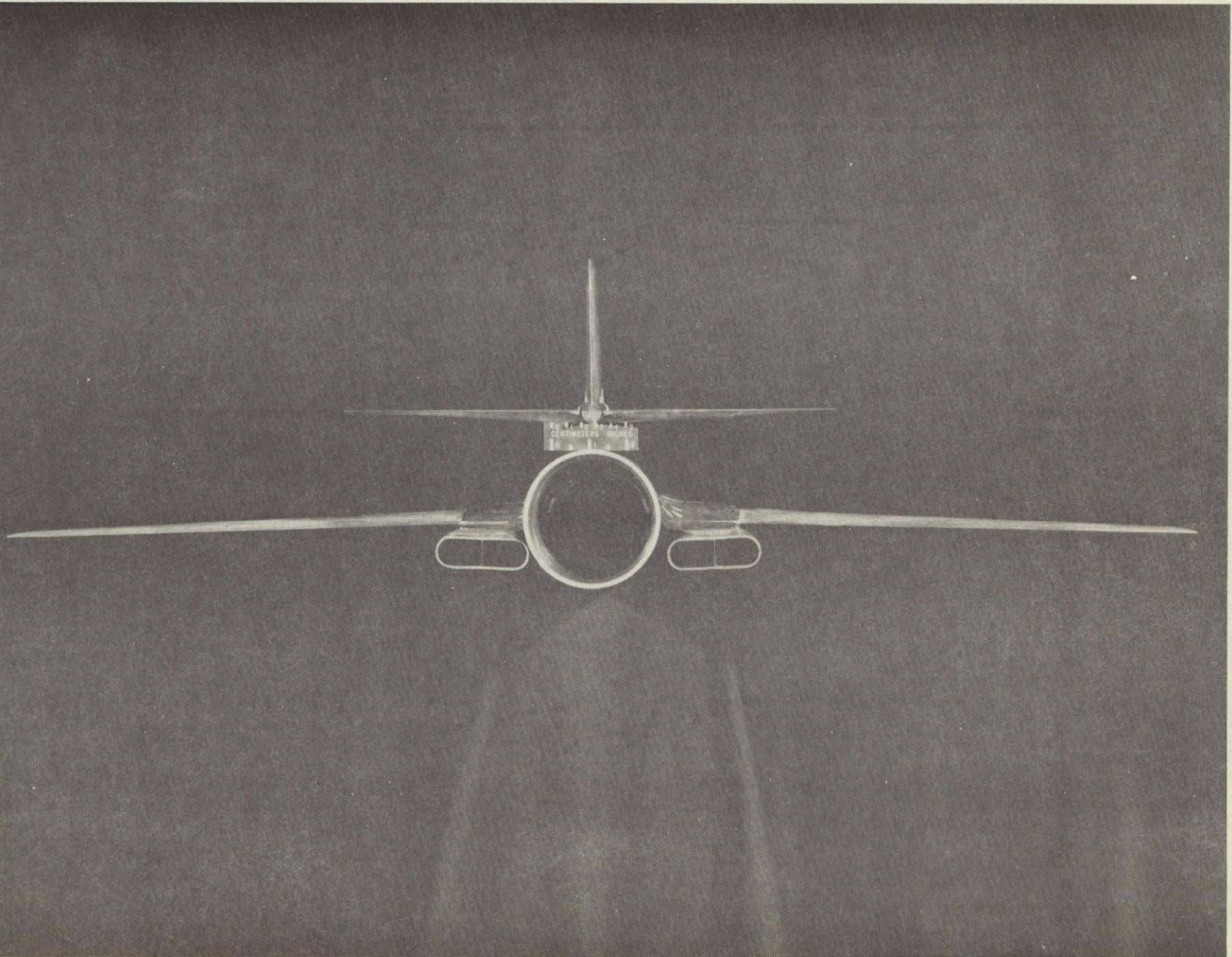
NASA
L-71-8370



0 10 20 30 40 50
CENTIMETERS INCHES

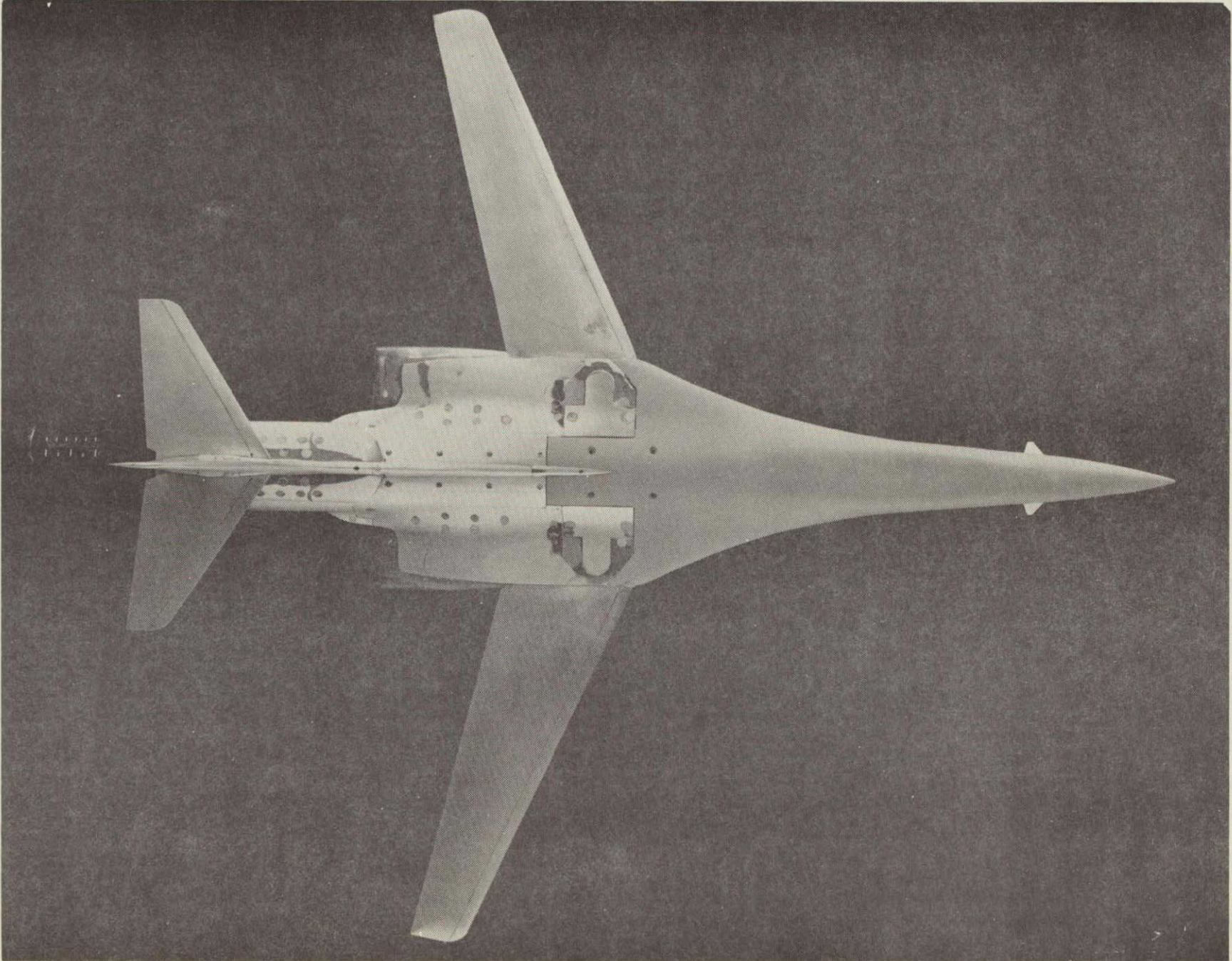
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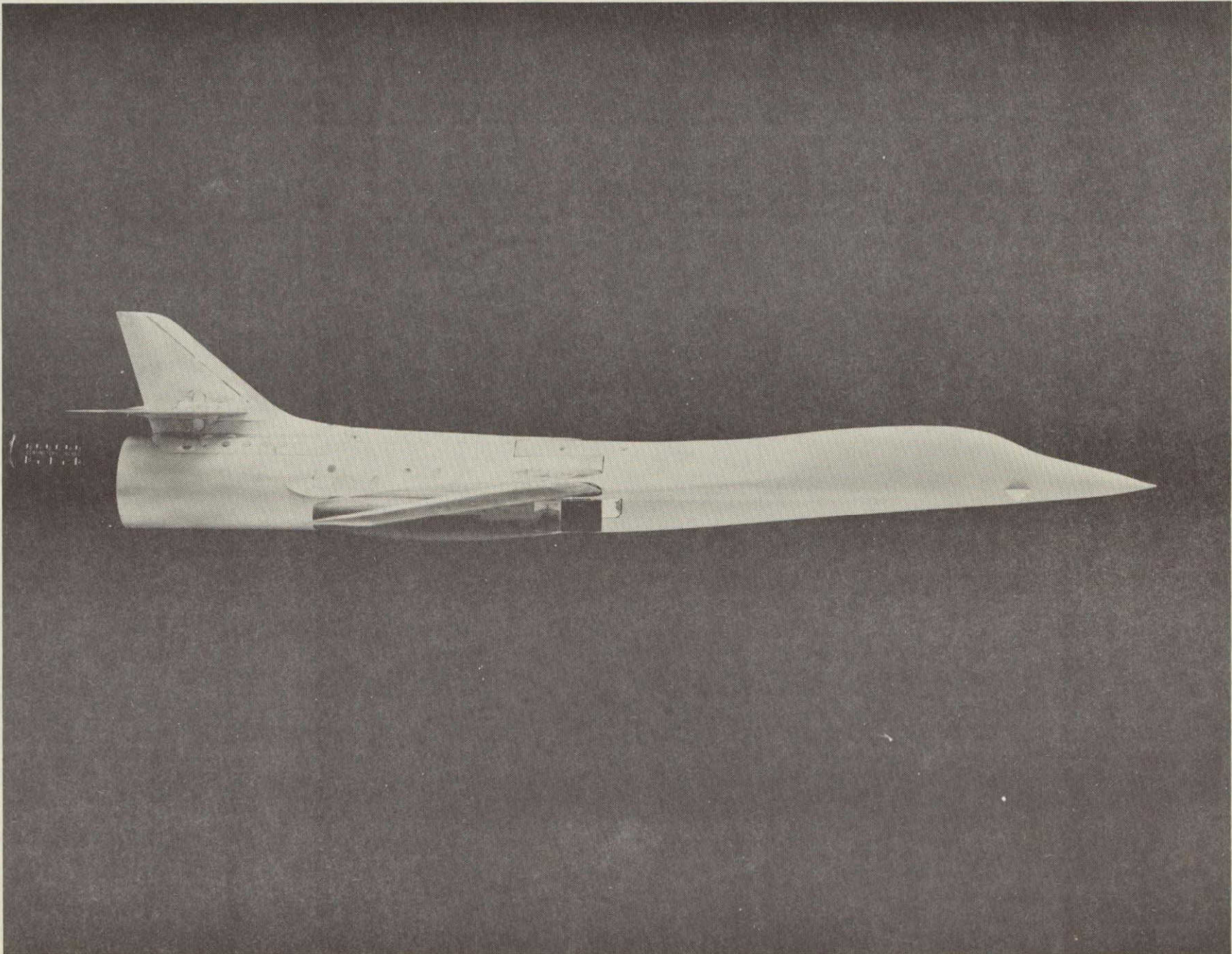
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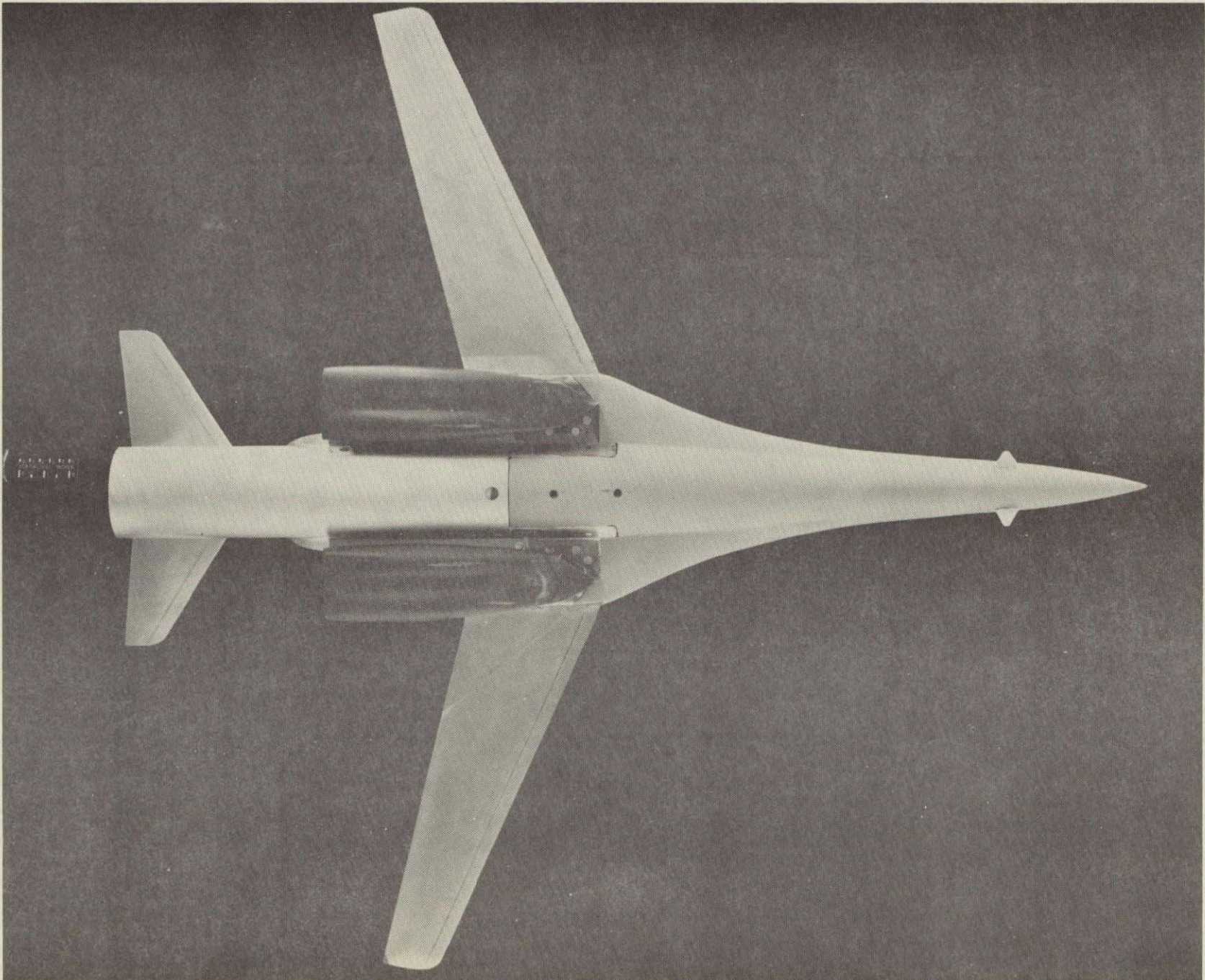
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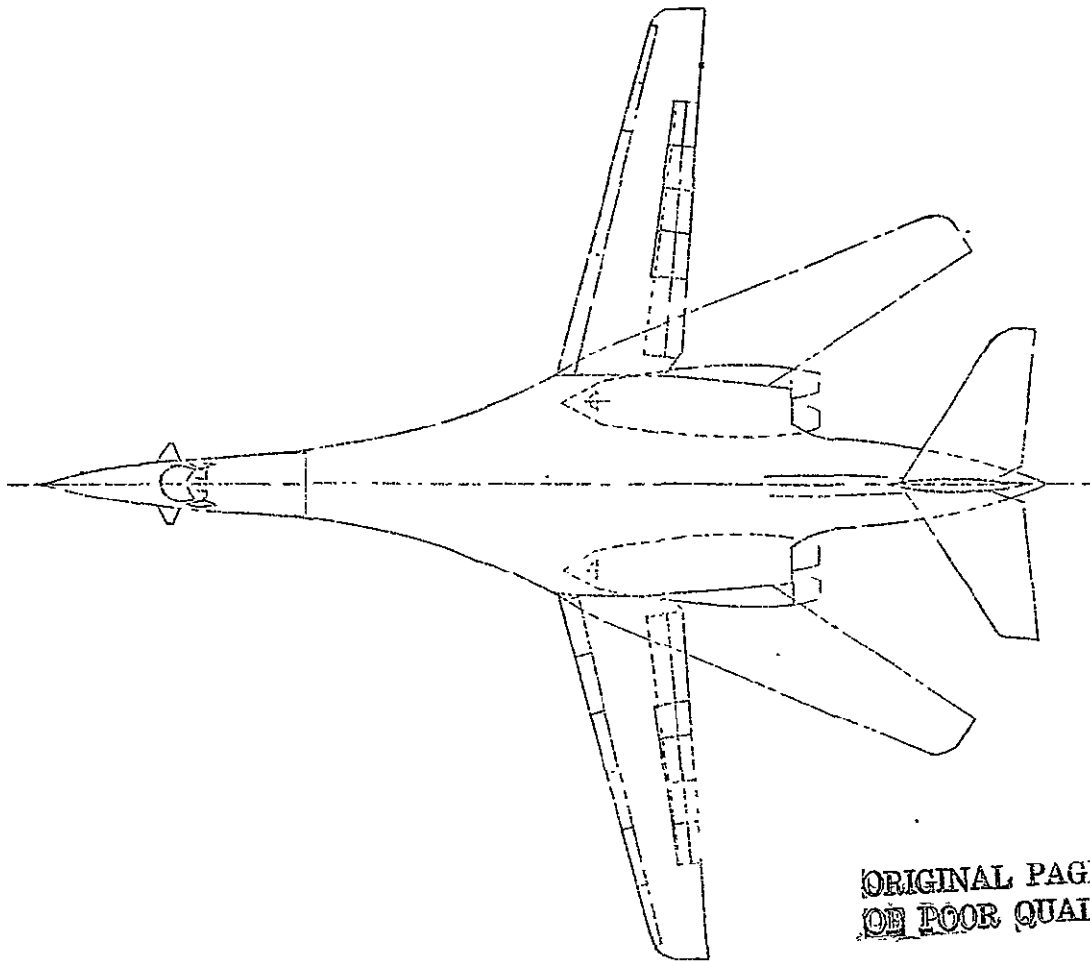


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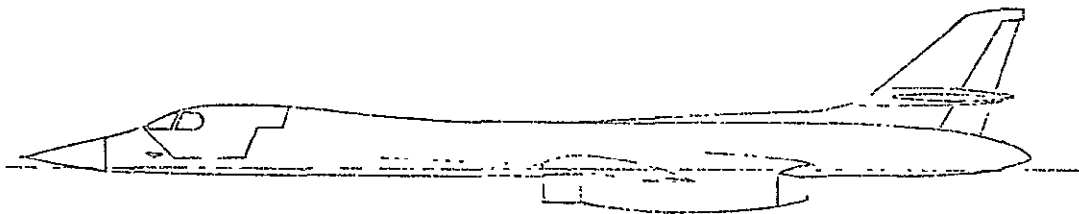
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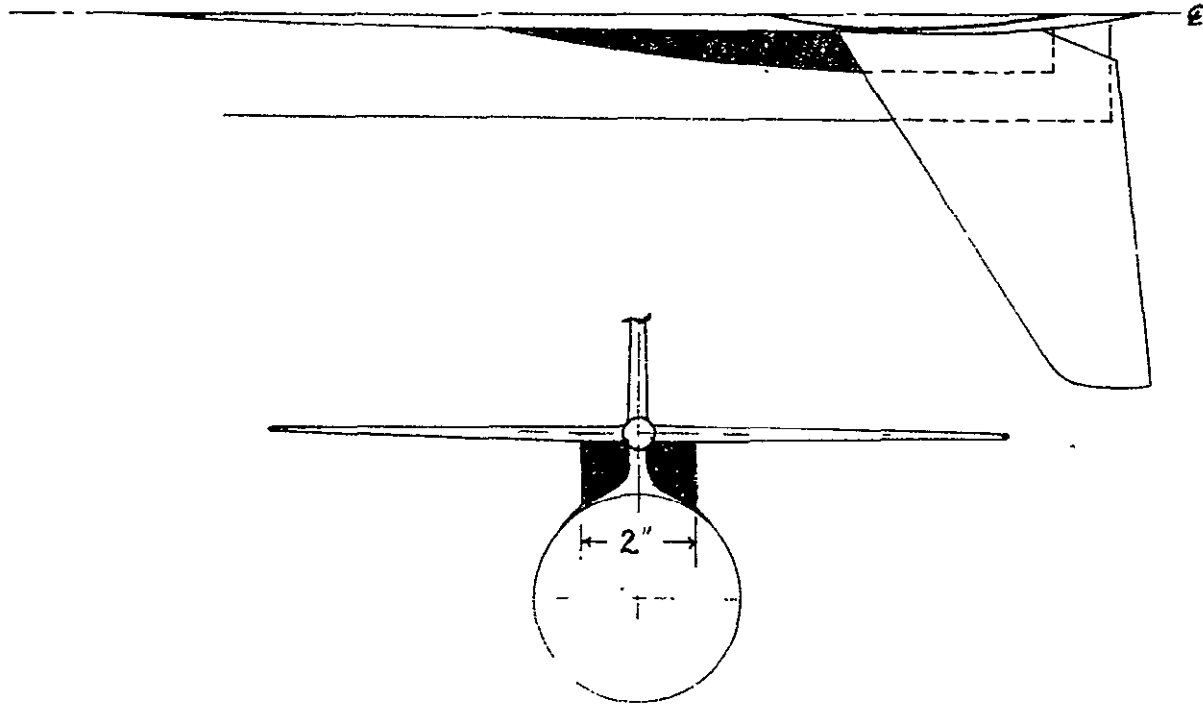


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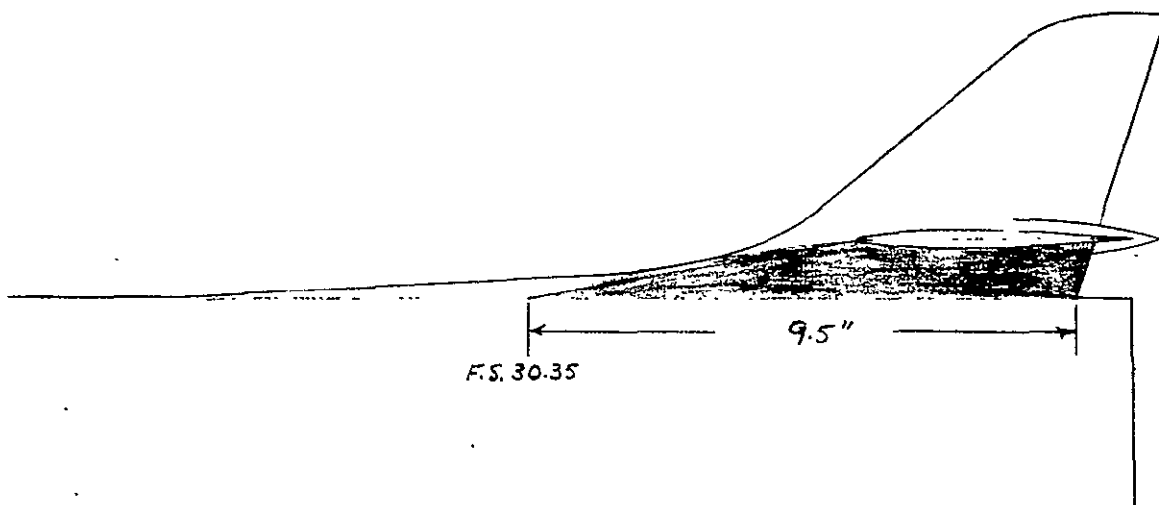


VIEW OF B-1

.024 SCALE ROTARY DERIVATIVE MODEL



LOOKING FORWARD



CHANNEL FILLER BLOCKS, B₆₁

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ORIGINAL PAGE 5
OF BOOK QUOTE

-30

-40

-50

-60

-70

-80

0

5

10

15

20

25

30

Mach NUMBER

EFFECT OF MACH NUMBER ON DRAGGING IN REACH

TALL ON WING $\alpha = 15^\circ$ $C_D = 25\%$

TEST NO. 570

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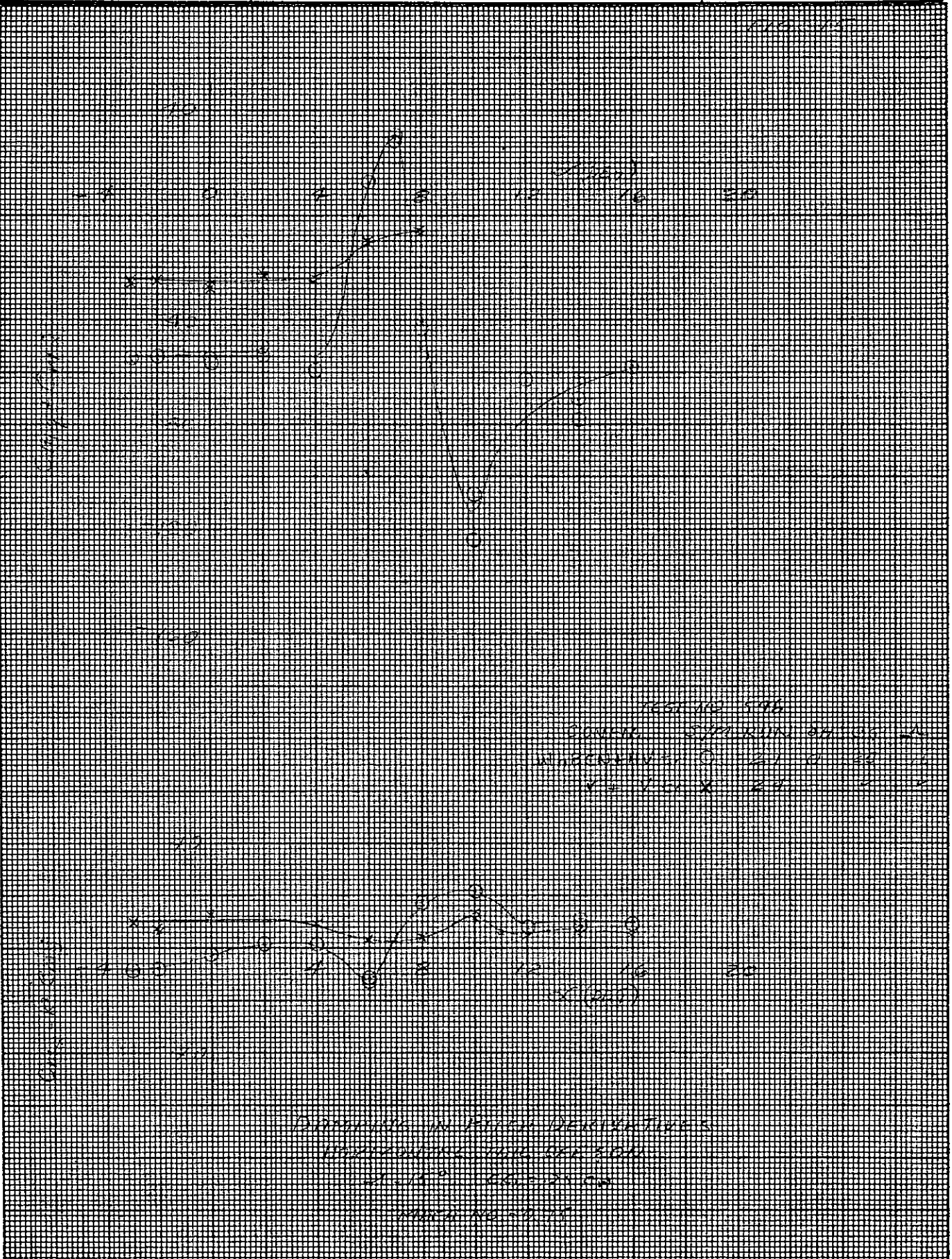
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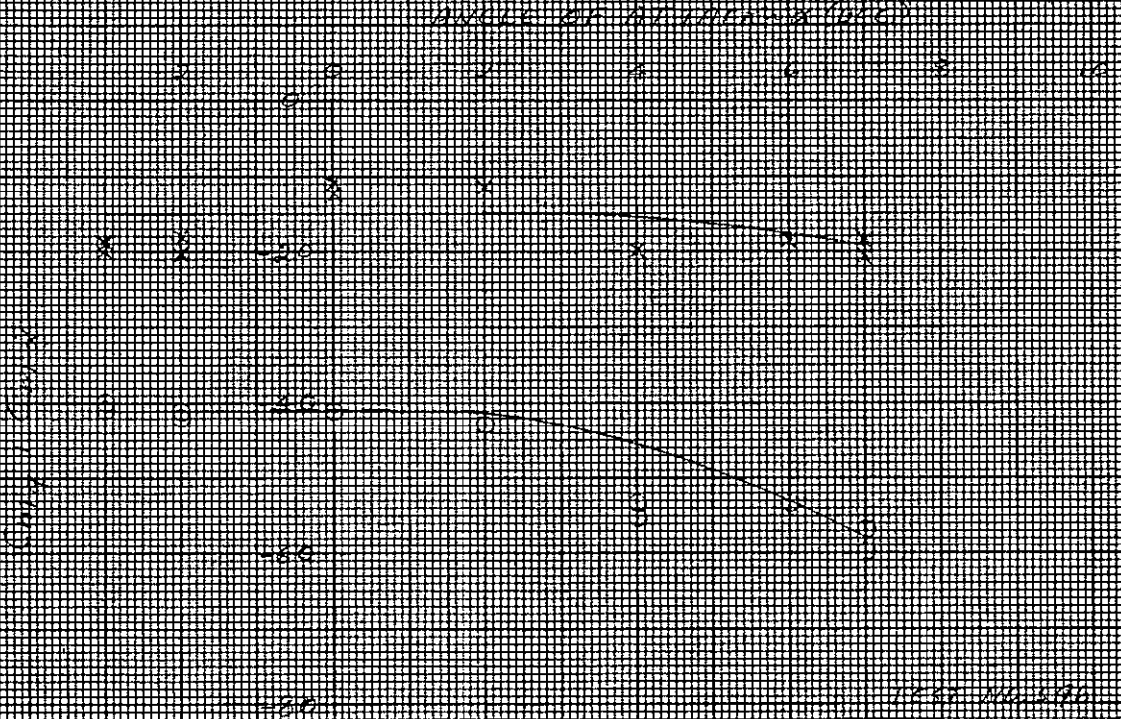
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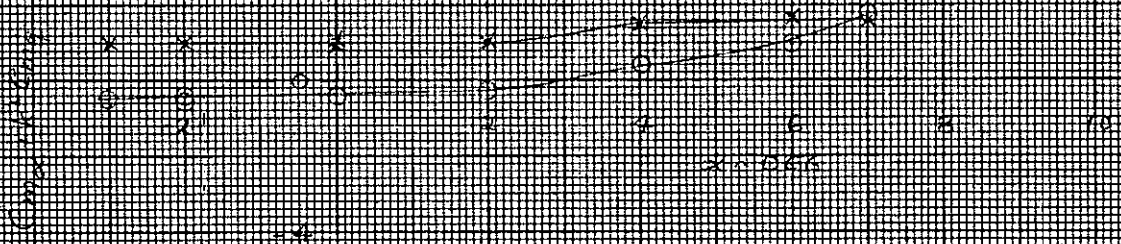
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MODEL NO.

FIG 16



TEST NUMBER
CIRCUIT 100 RUN 500000 20
WIND TUNNEL 0 10 OF 48 25
WIND TUNNEL 20 10 10



DATA IN FIRST REVISION
PORTABLE TAIL OFF 500
CL 20 80 400
MODEL NO 655

PREPARED BY: C. L. B.



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DATE: 11-5-71

MODEL NO

$$C_1 = 70, C_2 = 45, C_3 = 40, C_4 = 40$$

2. $C \sim WZ^2 \sim W^2 \sim W^2 \sim W^2$

4. 日 期: 2019 年 6 月 14 日

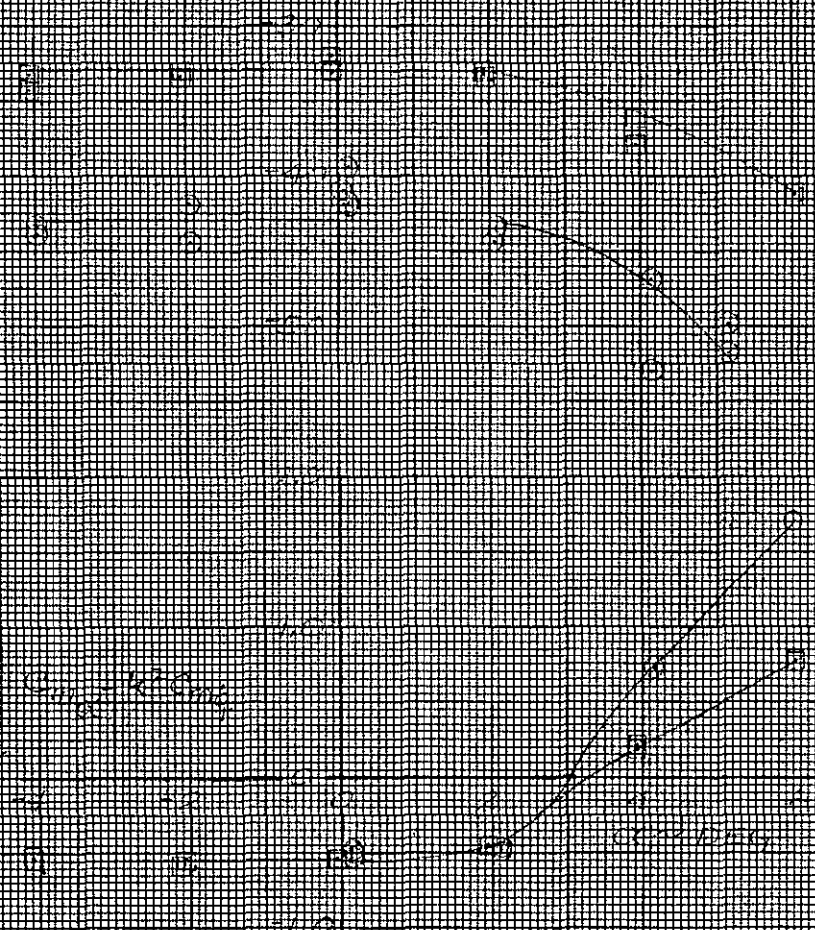
[illegible]

1000

Chemical

總發行所 東京 日本橋區本町二丁目三番地
 電話 二五八五
 支店 大阪 西區南船場四丁目一丁目
 電話 二五八五
 支店 京都 下京區東堀江二丁目一丁目
 電話 二五八五

EXP. DATA WITH LINEAR FITTING OF LINEAR
TRANSFORMED DATA



EFFECTS OF PESTICIDES ON MARSH AND COASTAL WILDLIFE

$\mu = 0.70$

REF NO: A36

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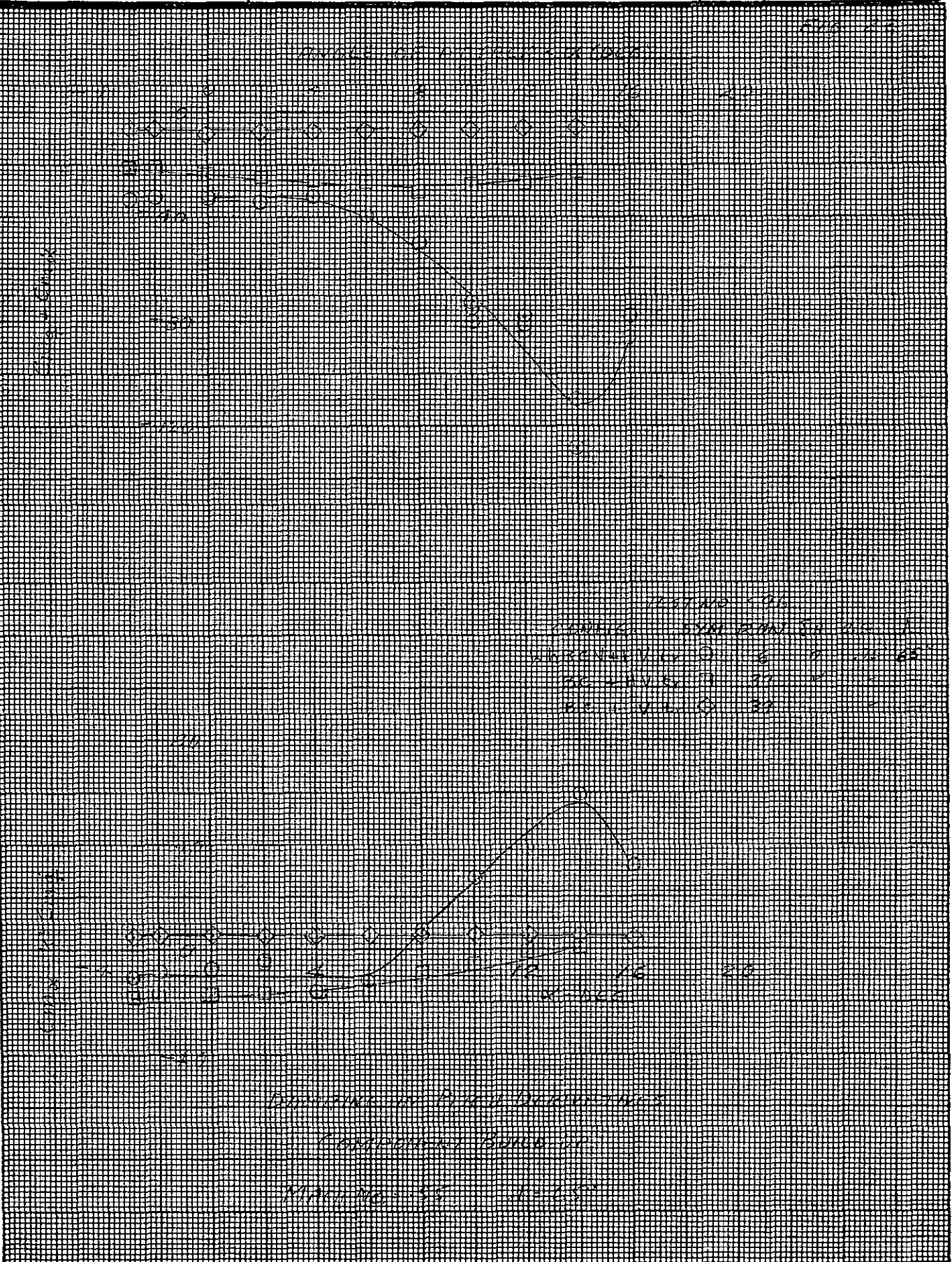


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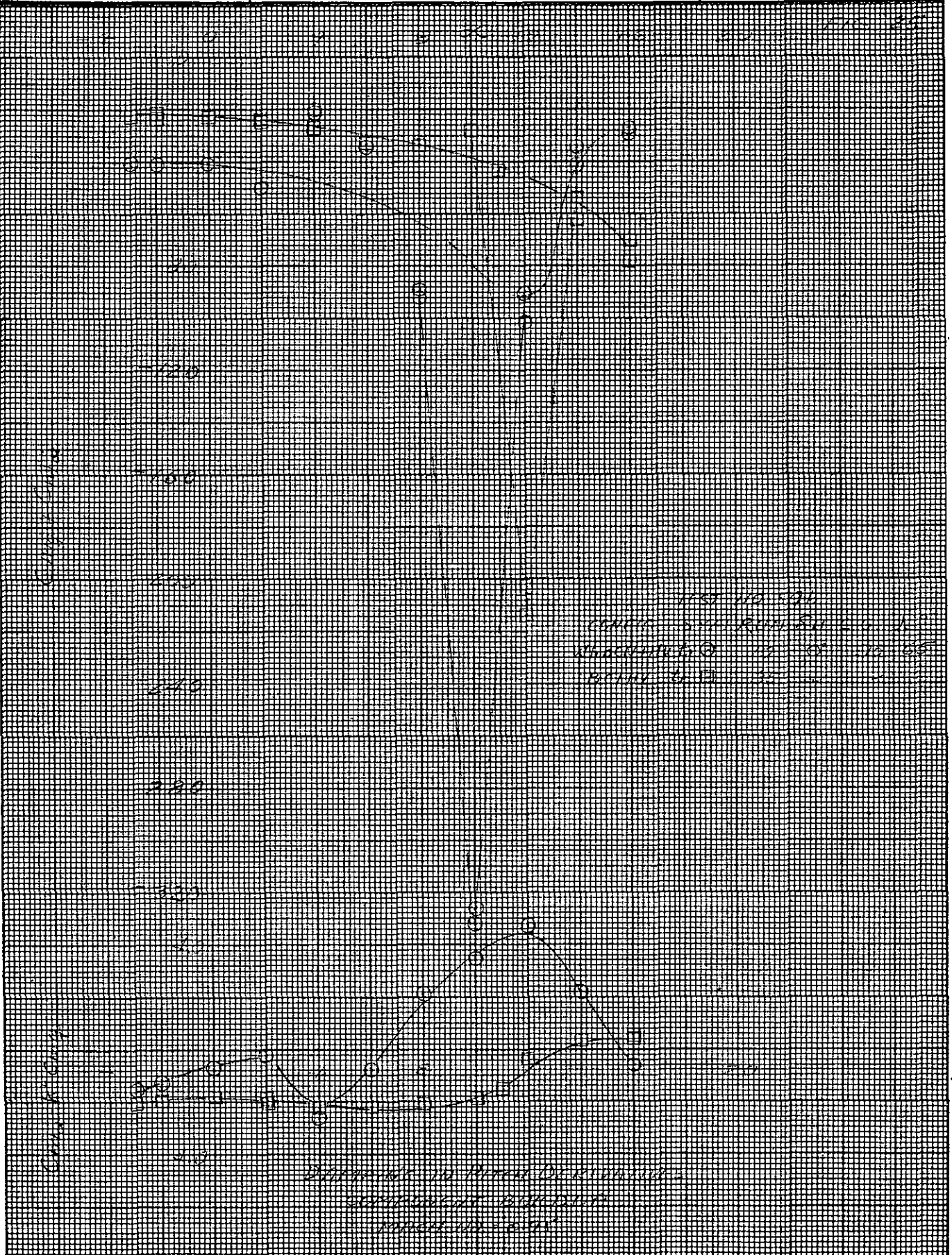
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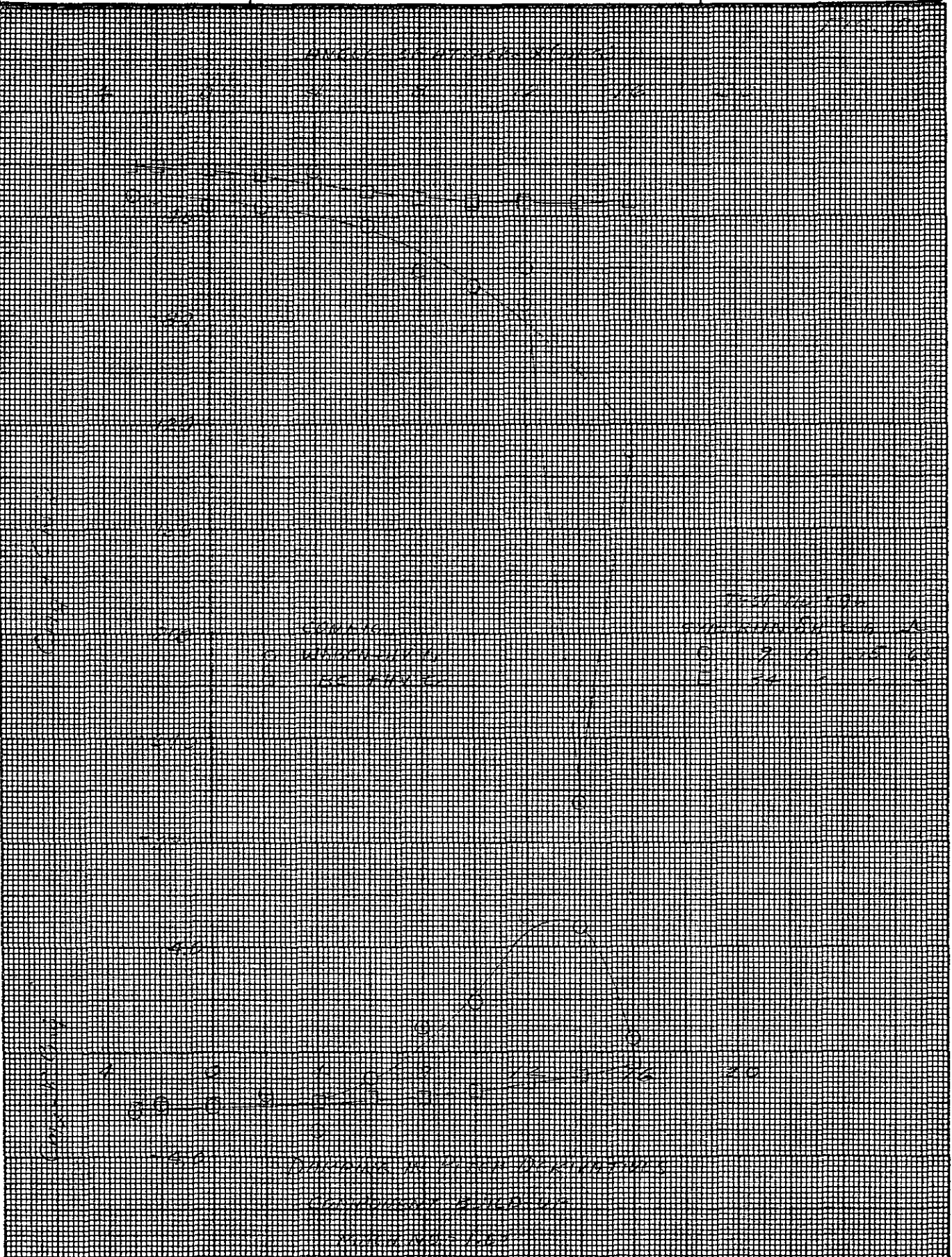
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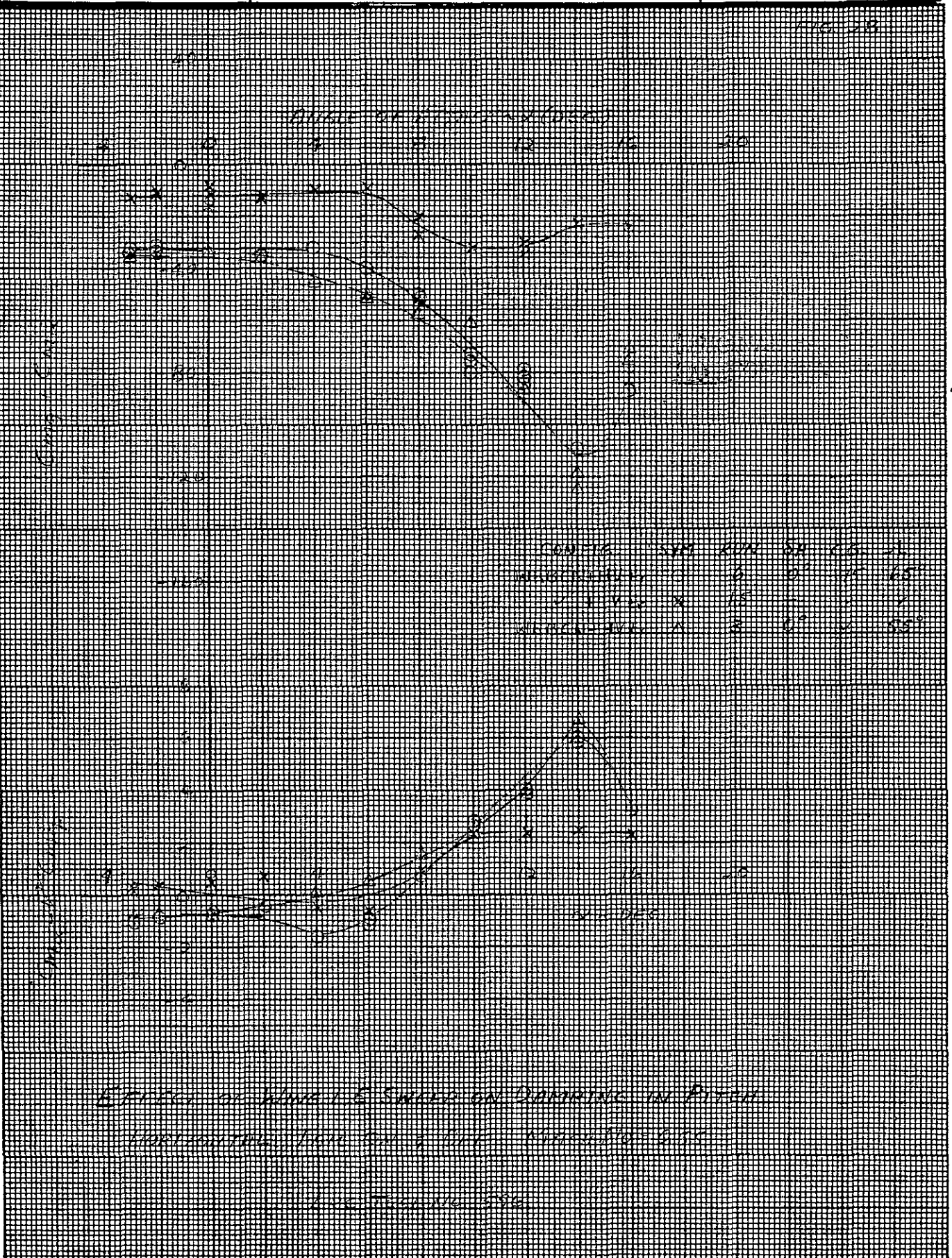
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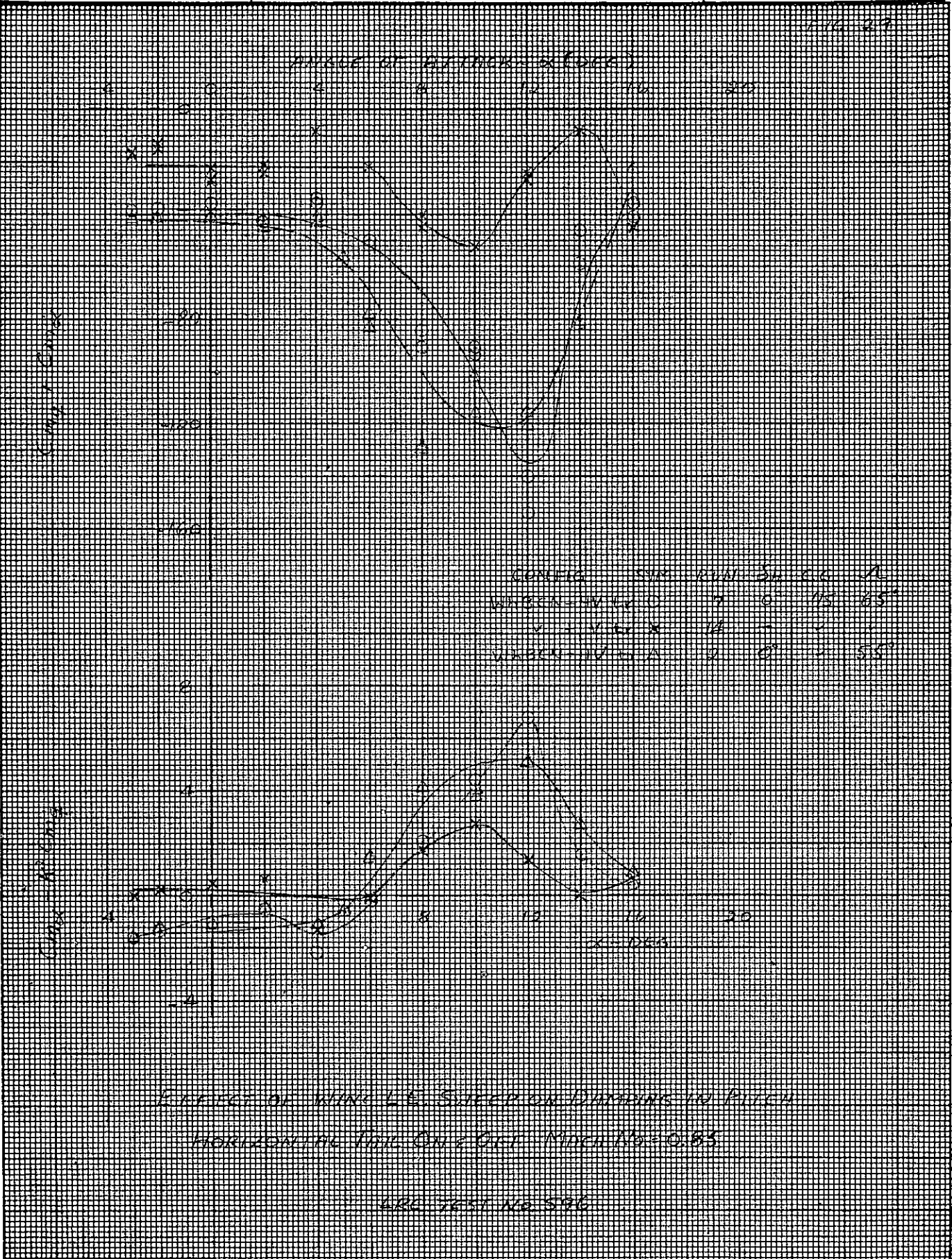
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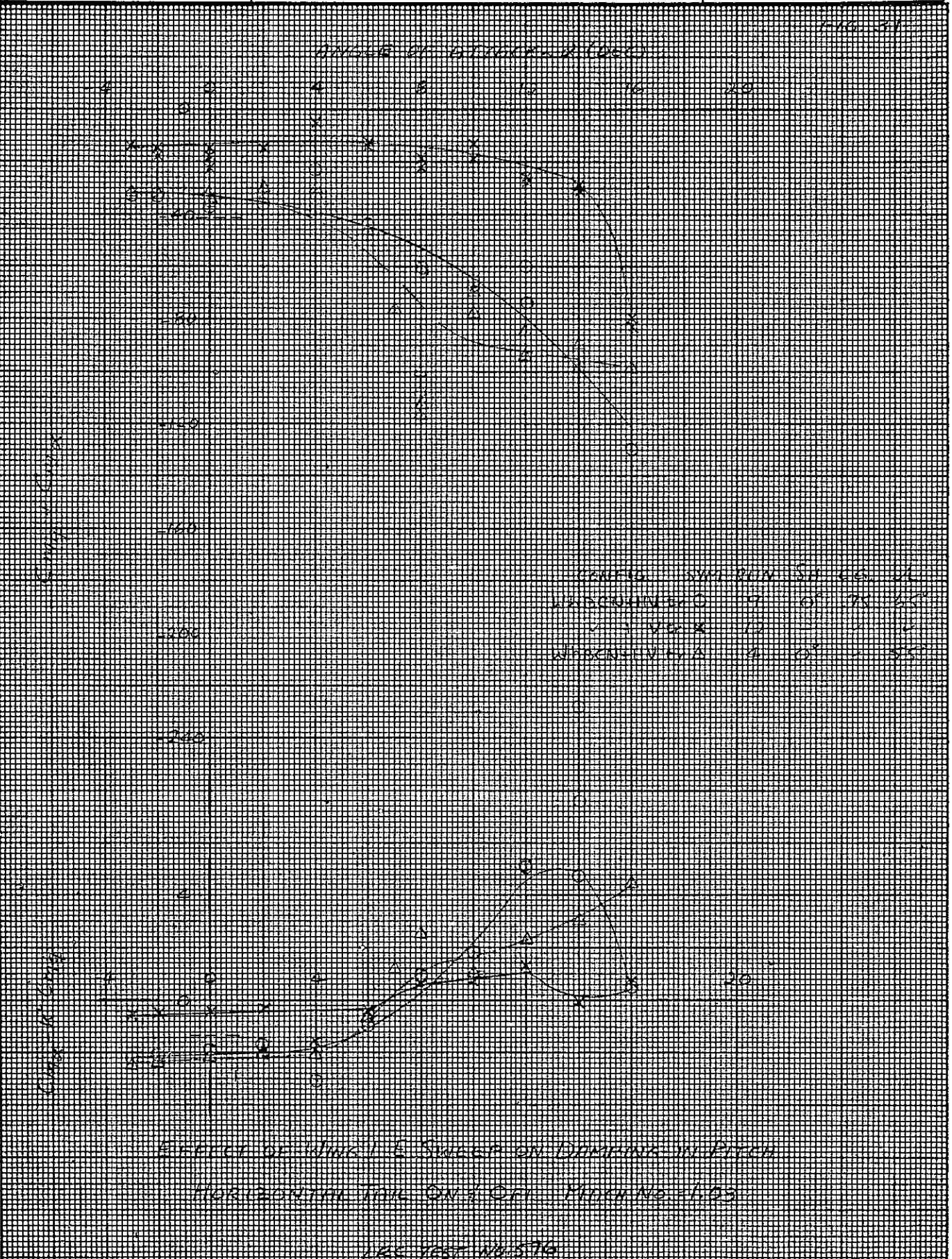
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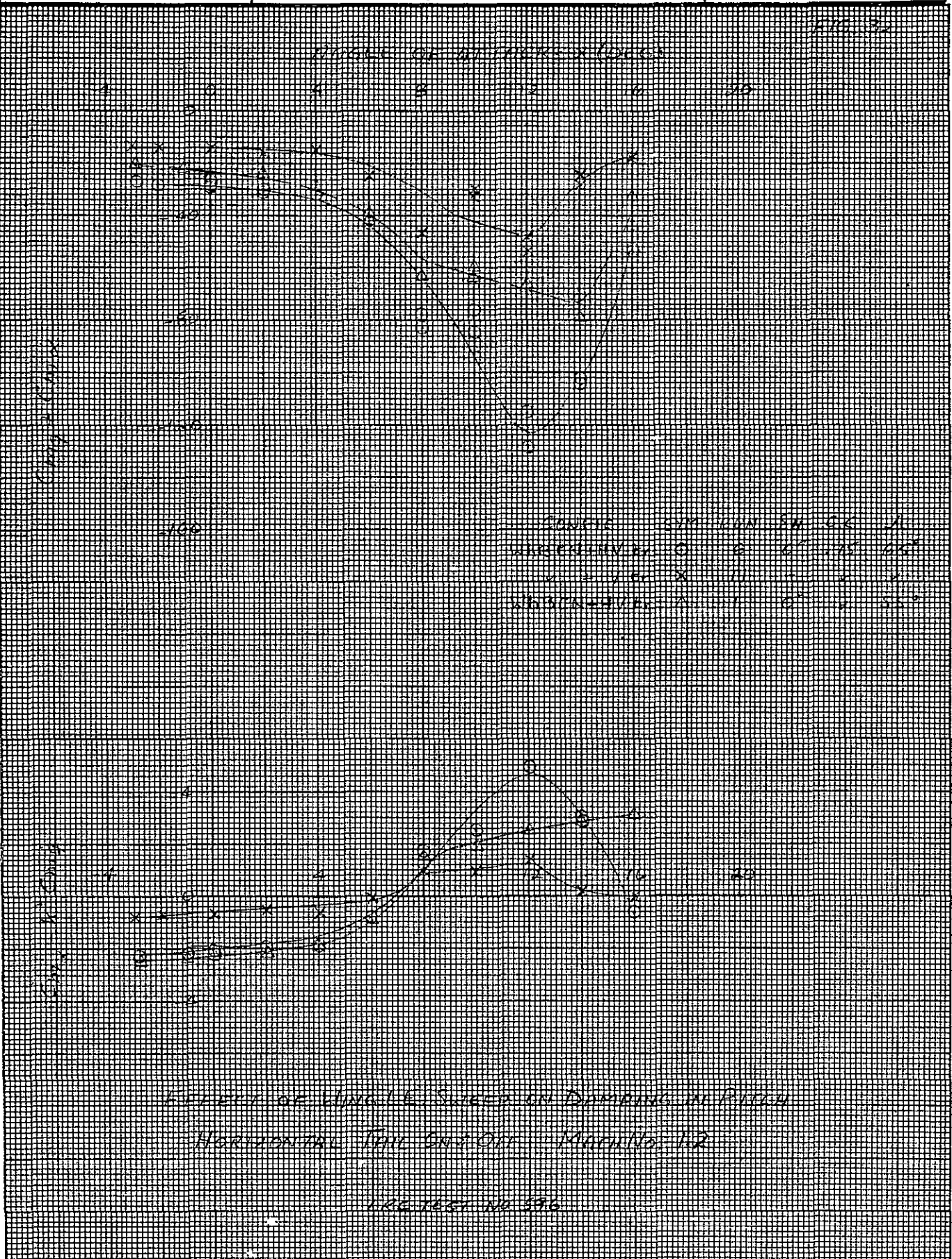
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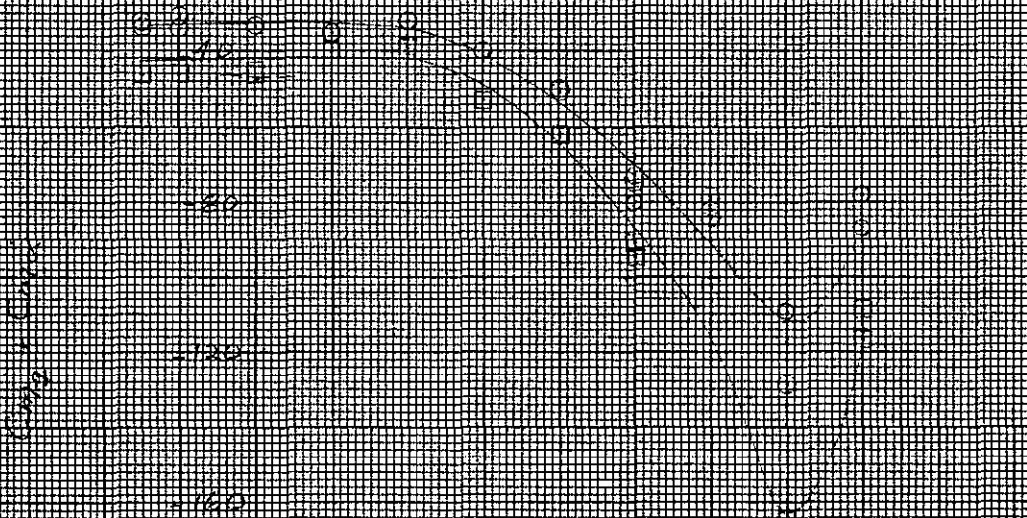
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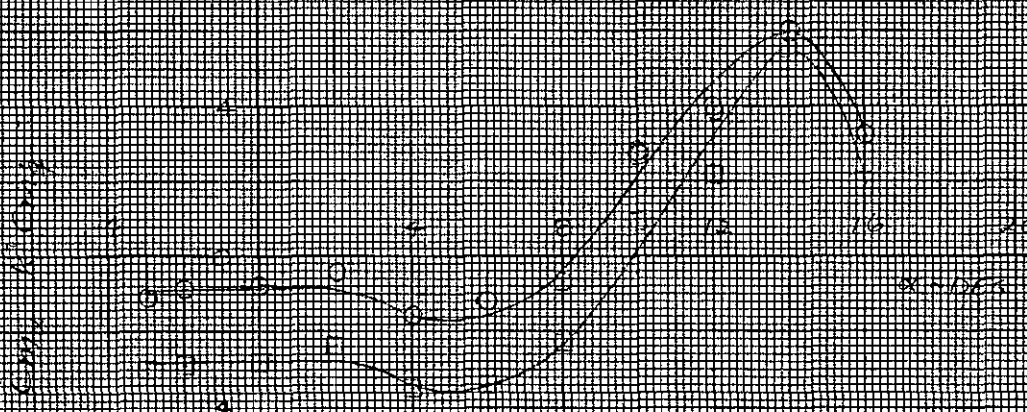
MODEL NO.

EFFECT OF ATTITUDE X (DEG)

4 6 8 10 12 14 16 18 20



SYN. ROLL SHA. 65 IN
WINGSPAN 0 6 12 18 24 30
X (DEG) 0 15 30 45 60 75 90



EFFECT OF CG LOCATION ON DARNING IN PITCH

HORIZONTAL TAIL ON MATR No 055

LAL TEST No 596

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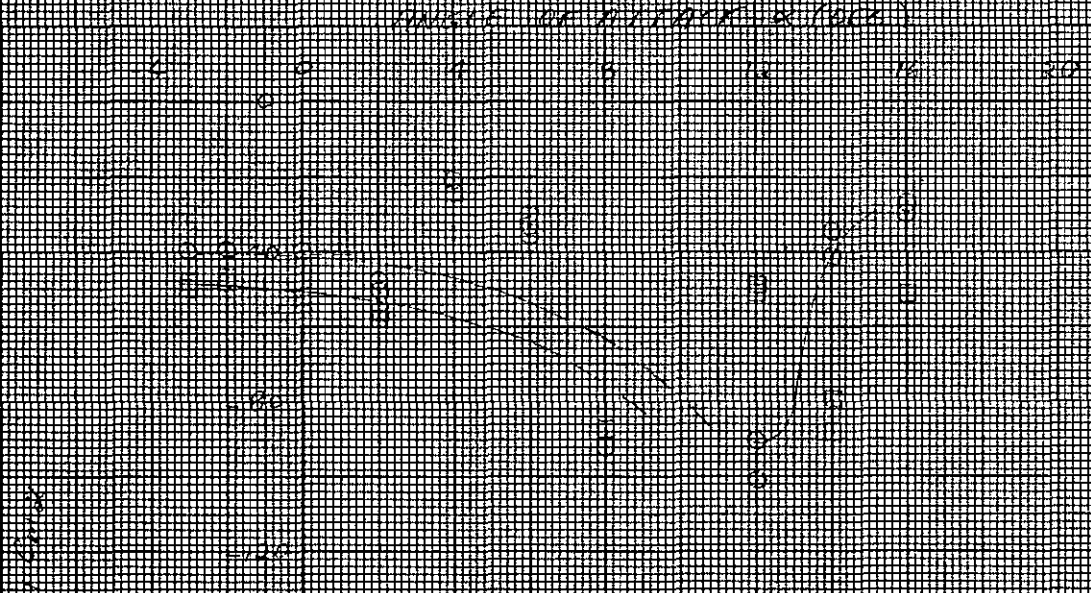
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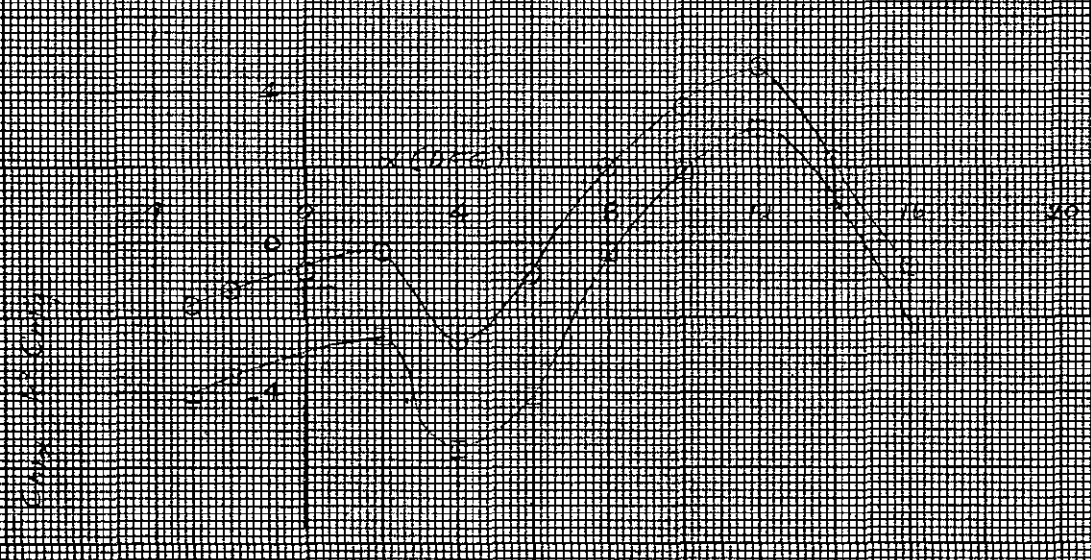
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MODEL NO


FIG. 35

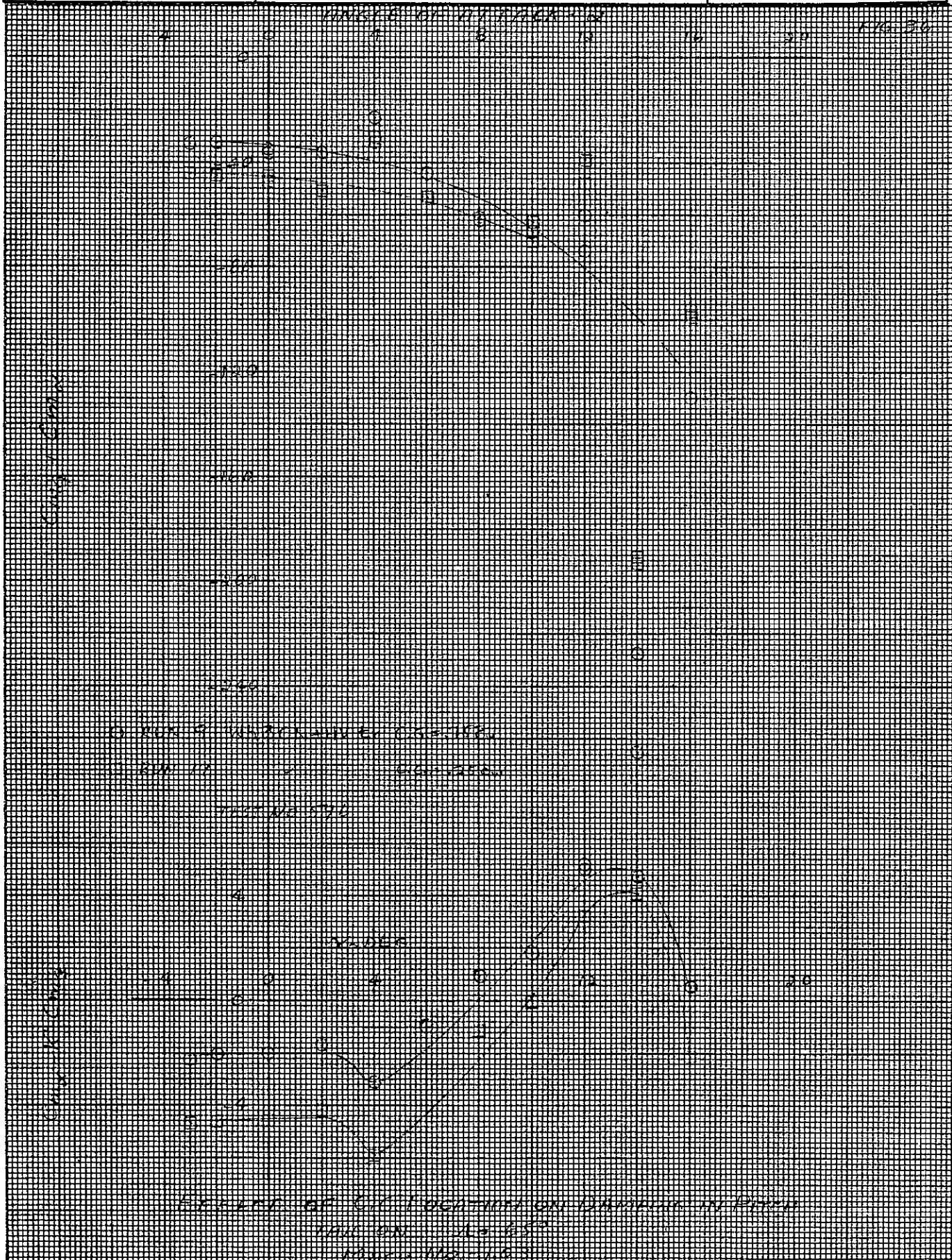



NOTE: 1. The magnitude of the response is a function of the frequency of the input. The magnitude of the response is a function of the frequency of the input. The magnitude of the response is a function of the frequency of the input.

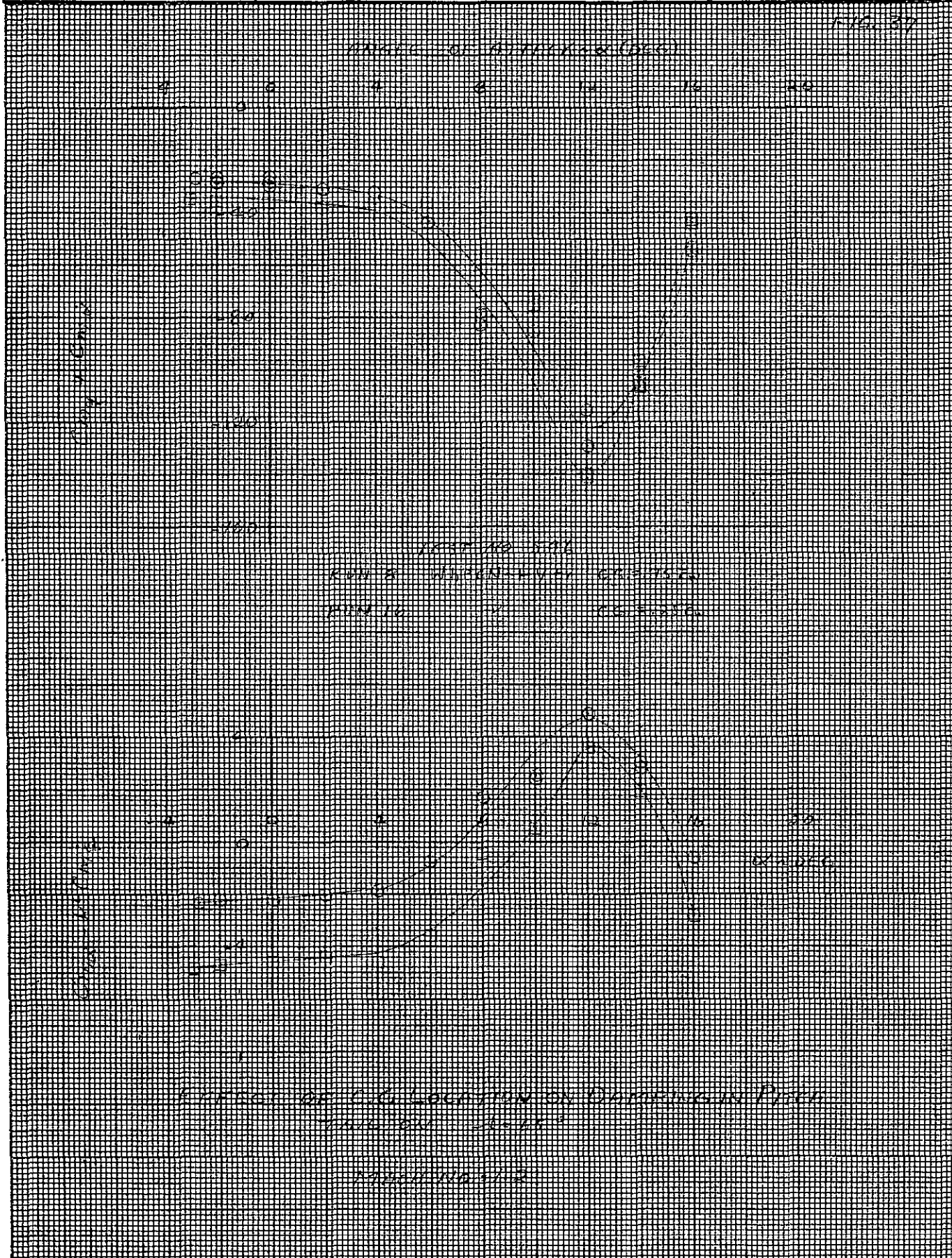


EFFECT OF COLLOCATION ON DAMPING IN RICH
SYSTEMS
PART NO. 596
LAC TEST NO. 596

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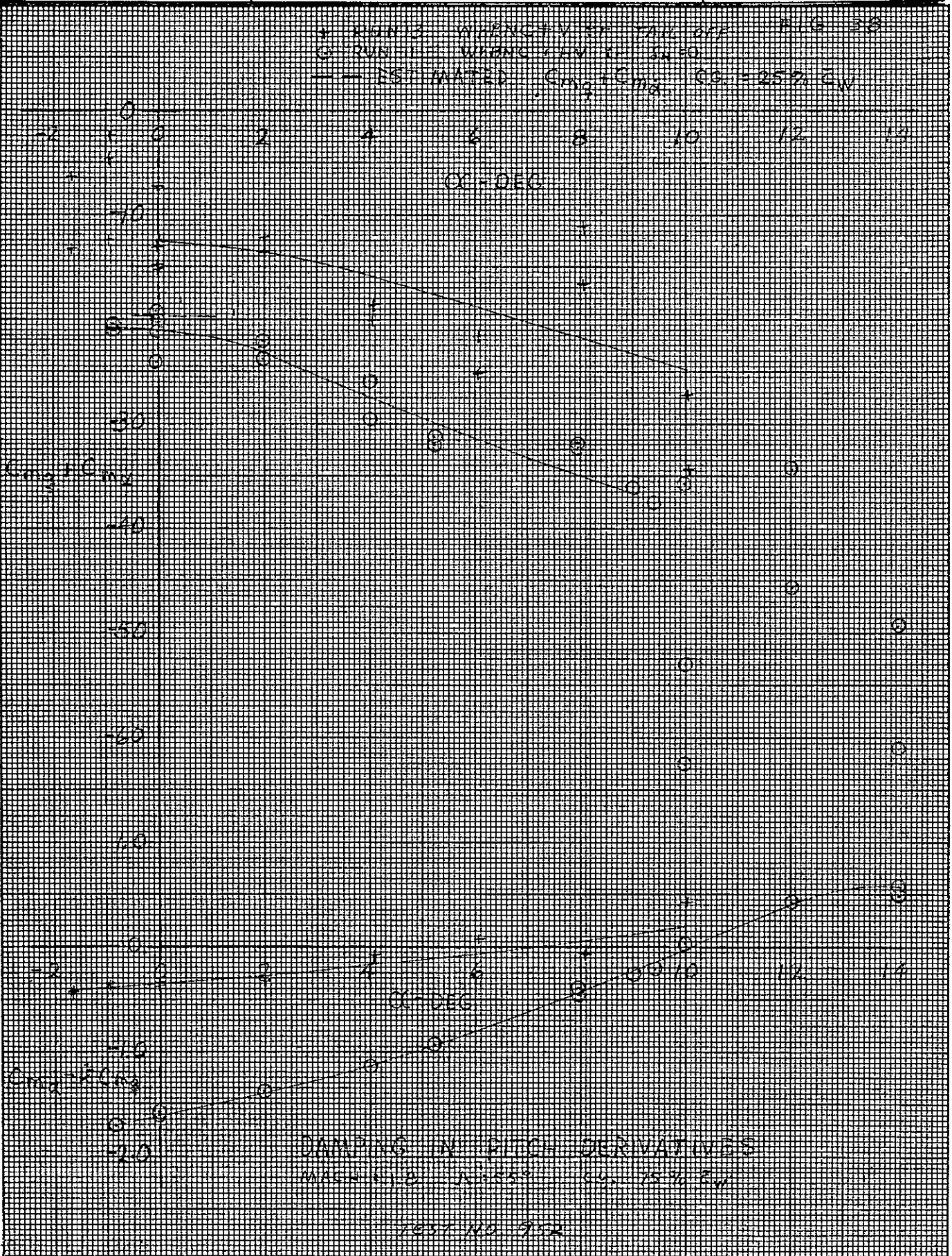
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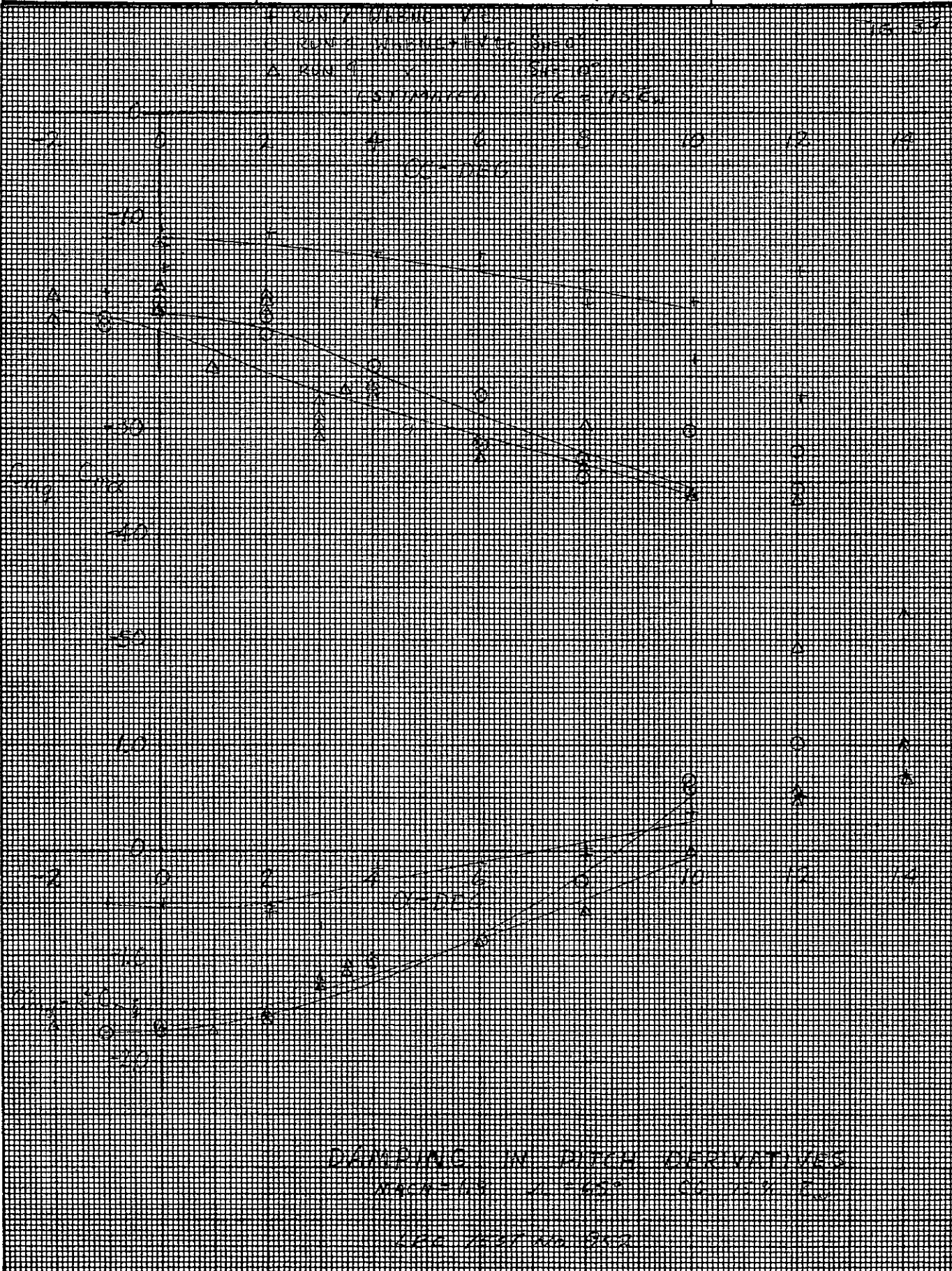
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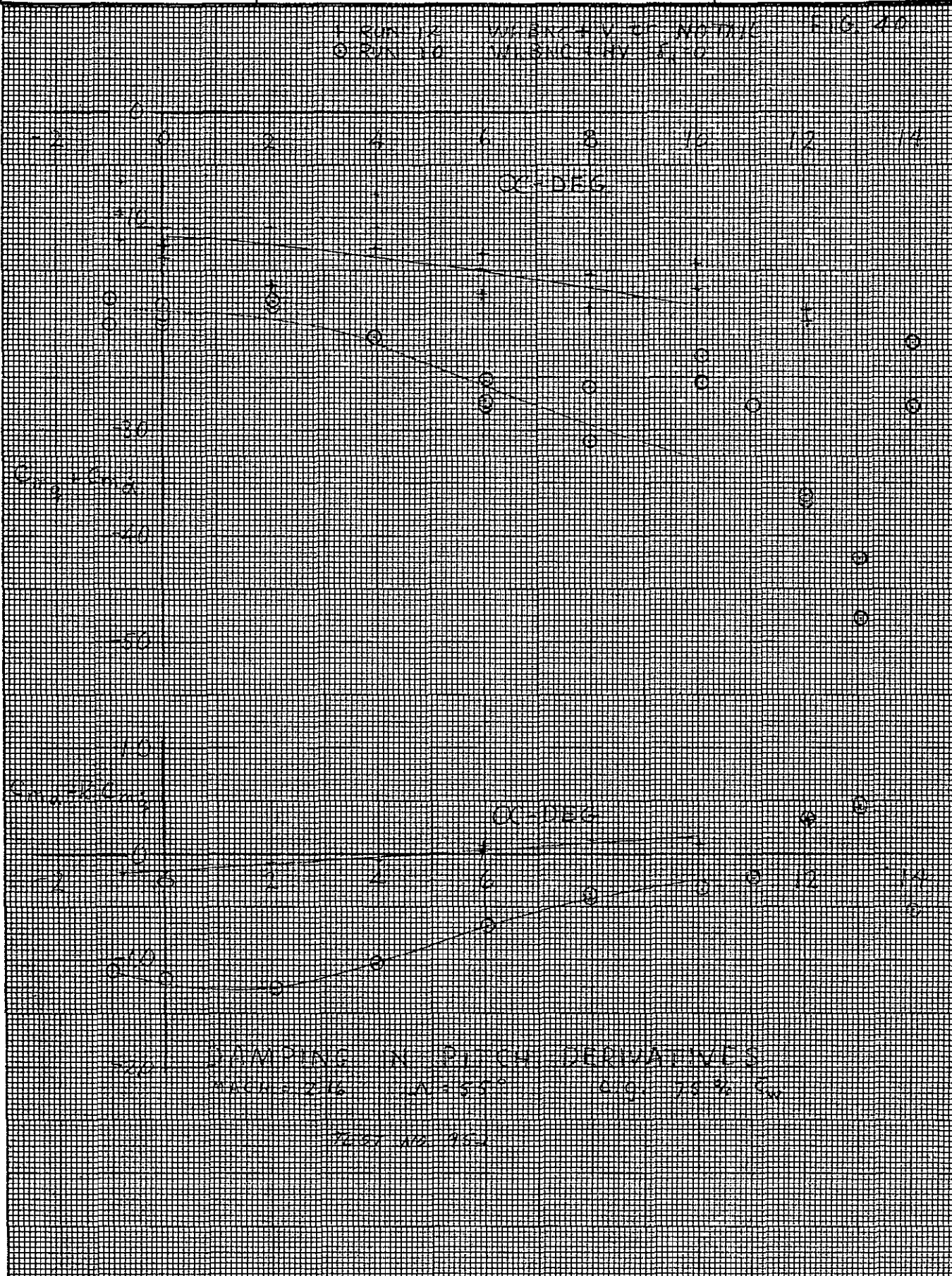
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1 RUN 6 WINDING V-5

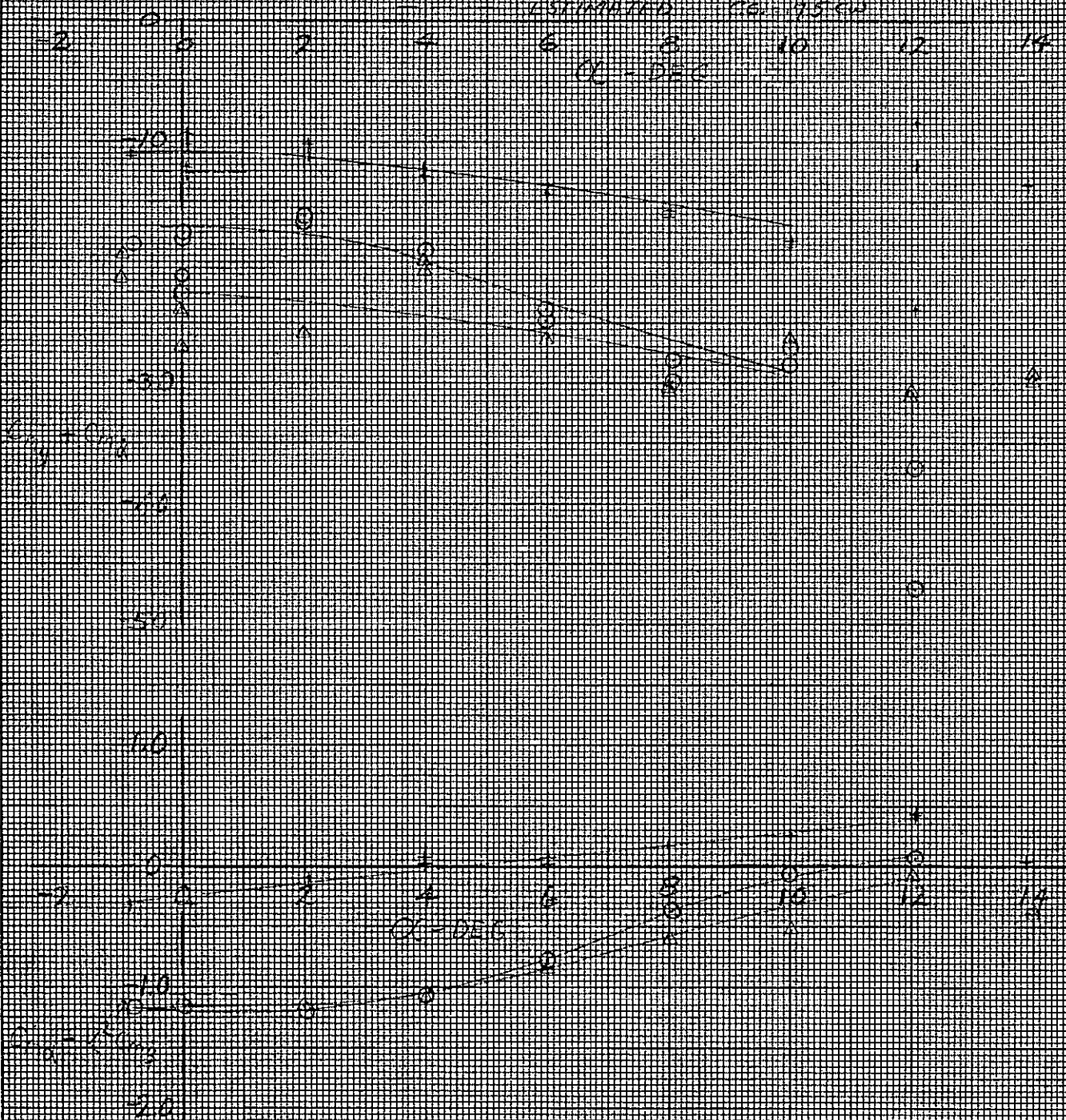
FIG. 21

2 RUN 8 WINDING V-5 SH=10°

3 RUN 3 WINDING V-5 SH=0°

ESTIMATED CG 75% CW

CL-DEC

DAMPING IN PITCH DERIVATIVES
WINDING V-5 SH=10° CG 75% CW

LRC TEST NO. 9K2

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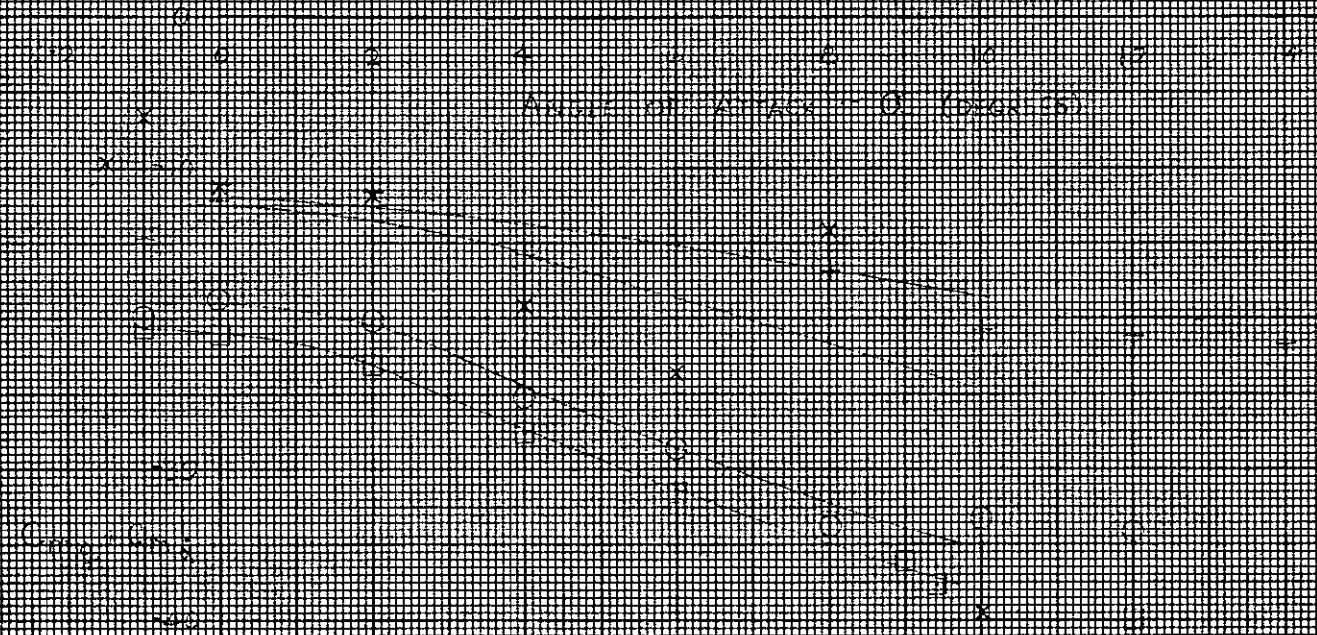
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MODEL NO.

O RUN 1 W/ENCLAVE A-165°
 + RUN 2 W/ENCLAVE A-165°
 E RUN 3 W/ENCLAVE A-165°
 X RUN 4 W/ENCLAVE A-165°



ORIGINAL DATA
OF DOOR ORAL

Drawn by: [Name] and [Name] in [Location] by [Name] Drawn by: [Name]

Scale: 1:1000

March 1972

NA-72-82

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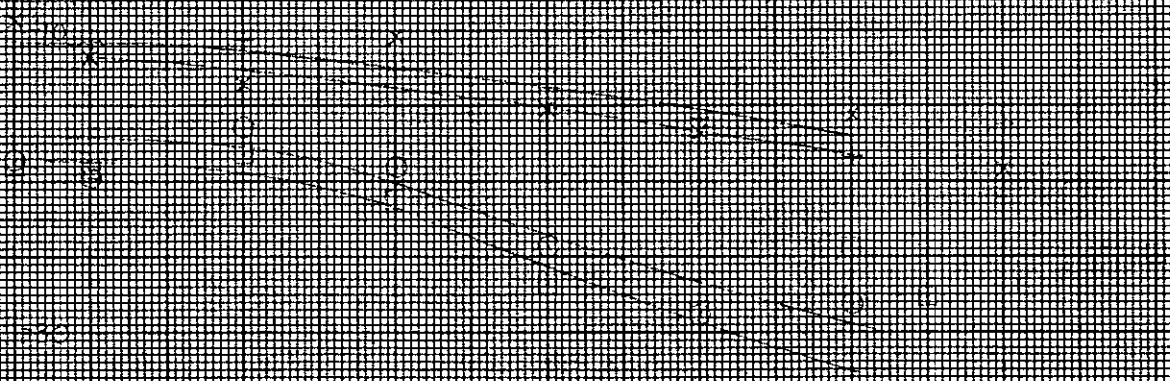
DATE:

MODEL NO.

510-15

O RUN-10 WASHOBY 8 A-555
P RUN-10 WASHOBY 10 A-555
R RUN-10 WASHOBY 8 A-555
X RUN-10 WASHOBY 8 A-555

1 2 3 4 5 6 7 8 9 10 11 12
WAVE OF ATTACK - (continued)



510-15

510-15

510-15

State of Washington - Bureau of State Development

510-15 510-15 510-15

WAVE NO. 1-15

WAVE NO. 1-15

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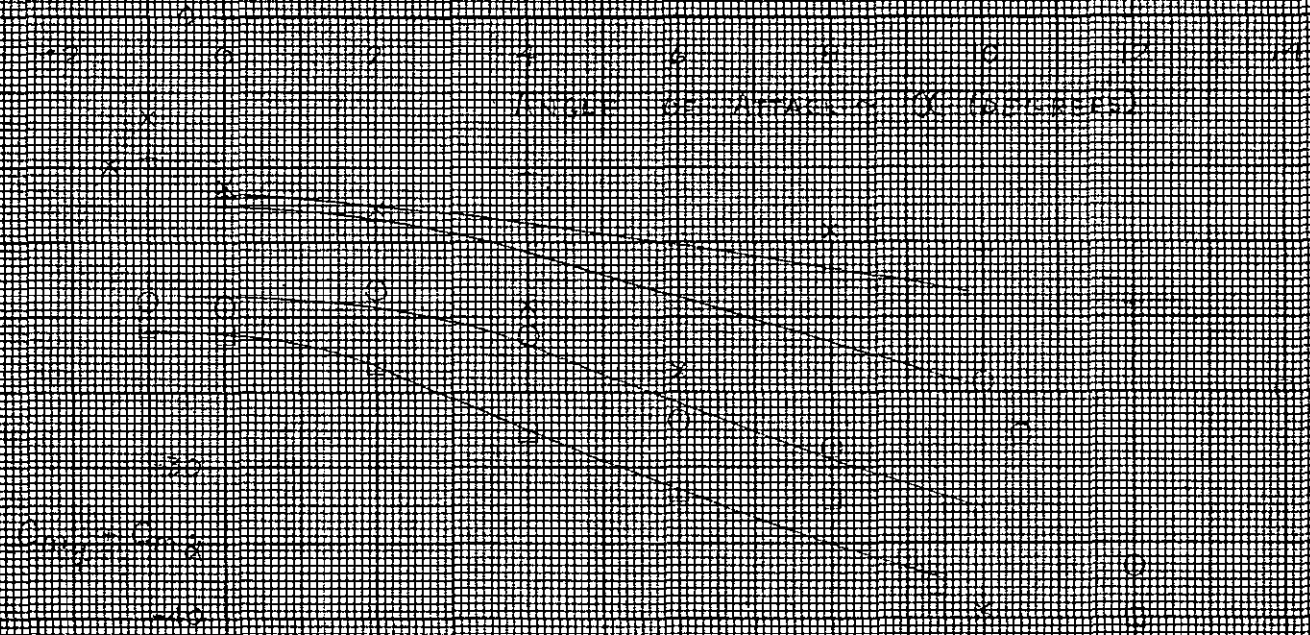
PAGE NO 70 OF 145

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MODEL NO

FIG 24

O RUN 10 WIND 1000 MARCH NO 12 10
 X RUN 11 WIND 1000 MARCH NO 13 10
 O RUN 12 WIND 1000 MARCH NO 14 10
 X RUN 13 WIND 1000 MARCH NO 15 10



EFFECT OF WIND TUNNEL INSTALLATION ON DRAG COEFFICIENT

WIND SPEED 1000 FT/SEC

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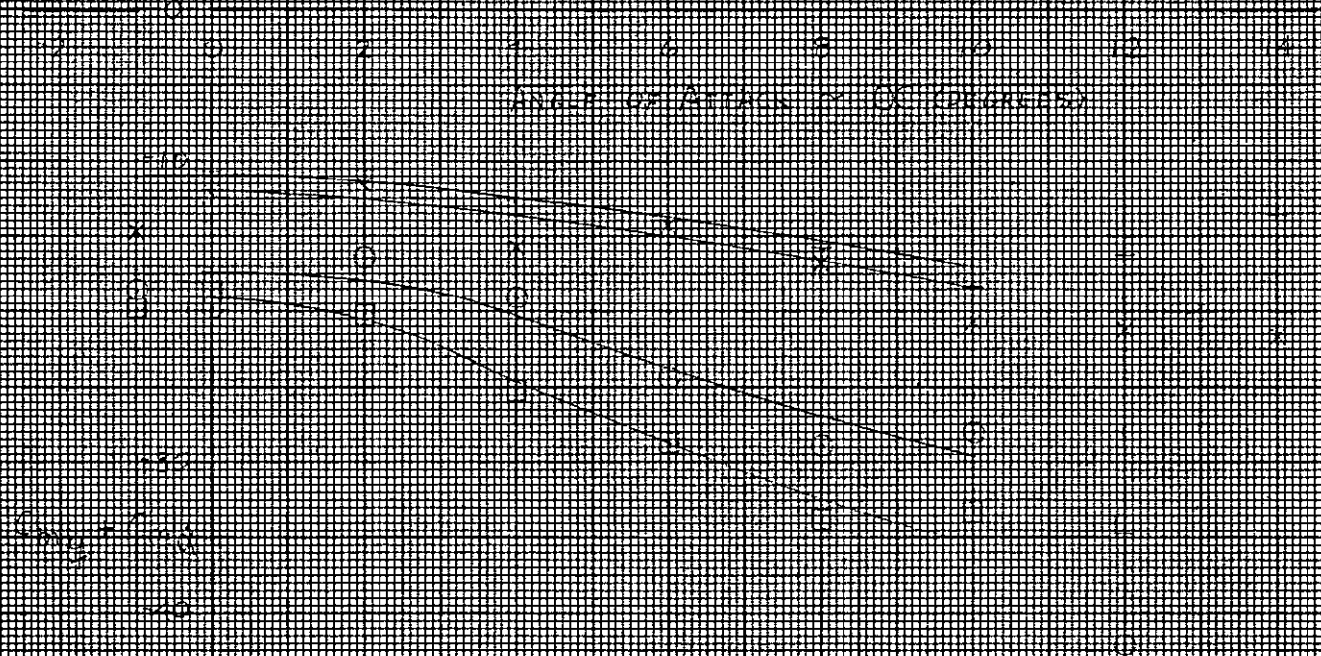
PAGE NO. 71 OF 145

REPORT NO. NA-72-82

MODEL NO

F. G. 15

- ☐ RUN 3 WINDING R. MATCH NO. 1216
- ☐ RUN 4 WINDING R. MATCH NO. 1220
- ☐ RUN 5 WINDING R. MATCH NO. 1216
- ☒ RUN 7 WINDING R. MATCH NO. 1220



EFFECT OF WASCOURR VARIATION ON DYNAMIC MOTOR DEMAND

WASCOURR = 1000

TEST NO. 700

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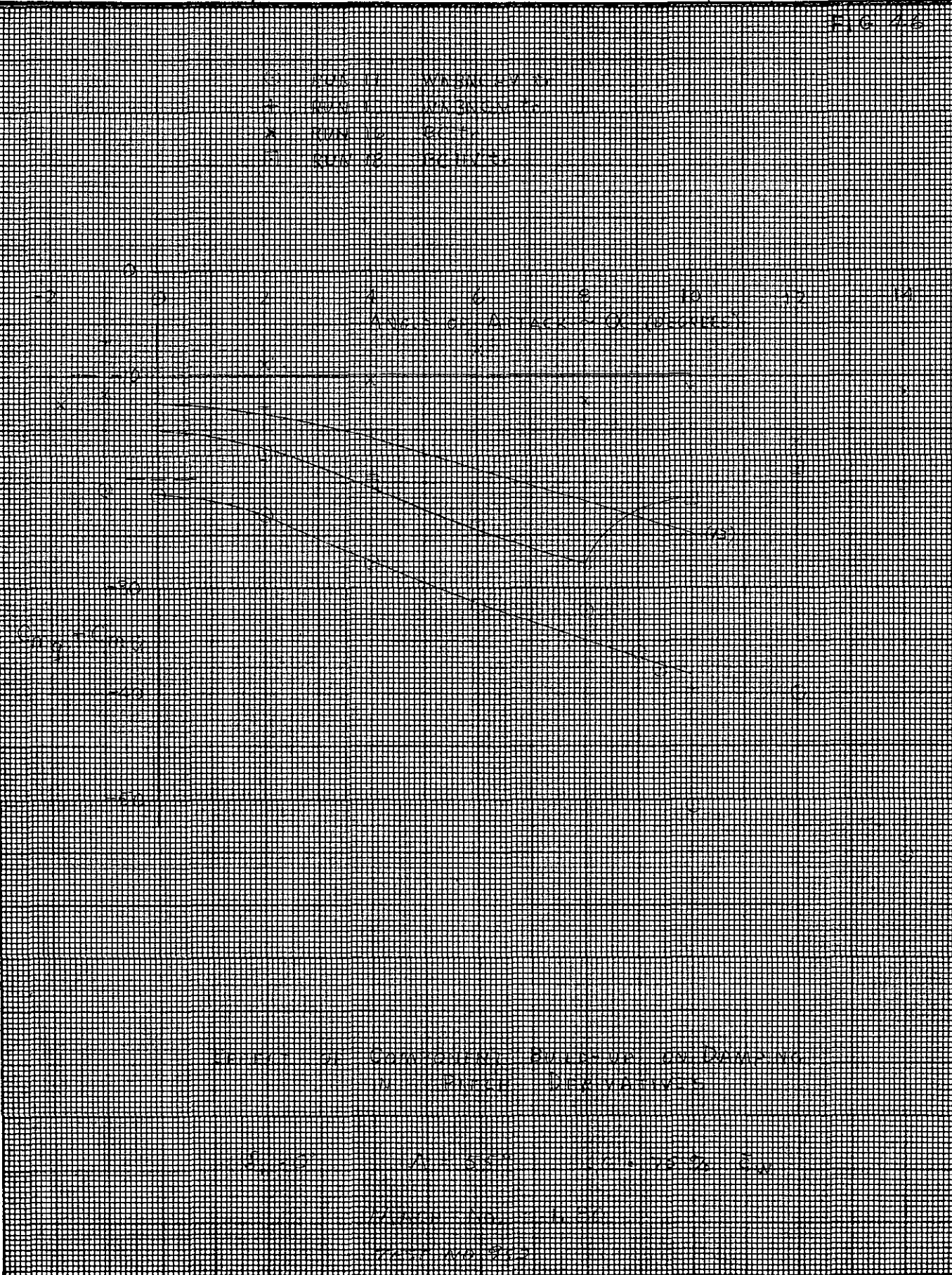
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MODEL NO.

FIG 48



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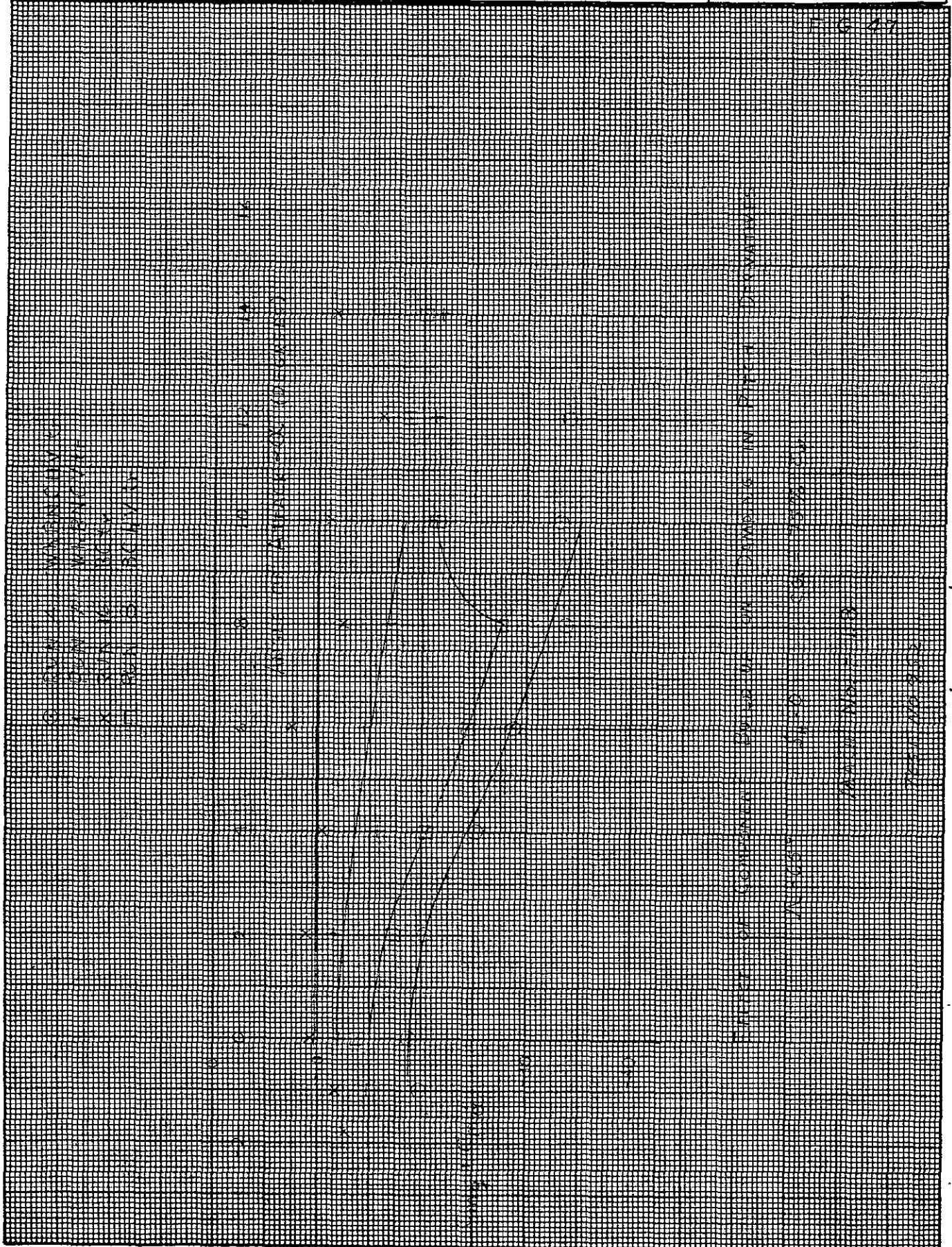
PAGE NO 73 OF 145

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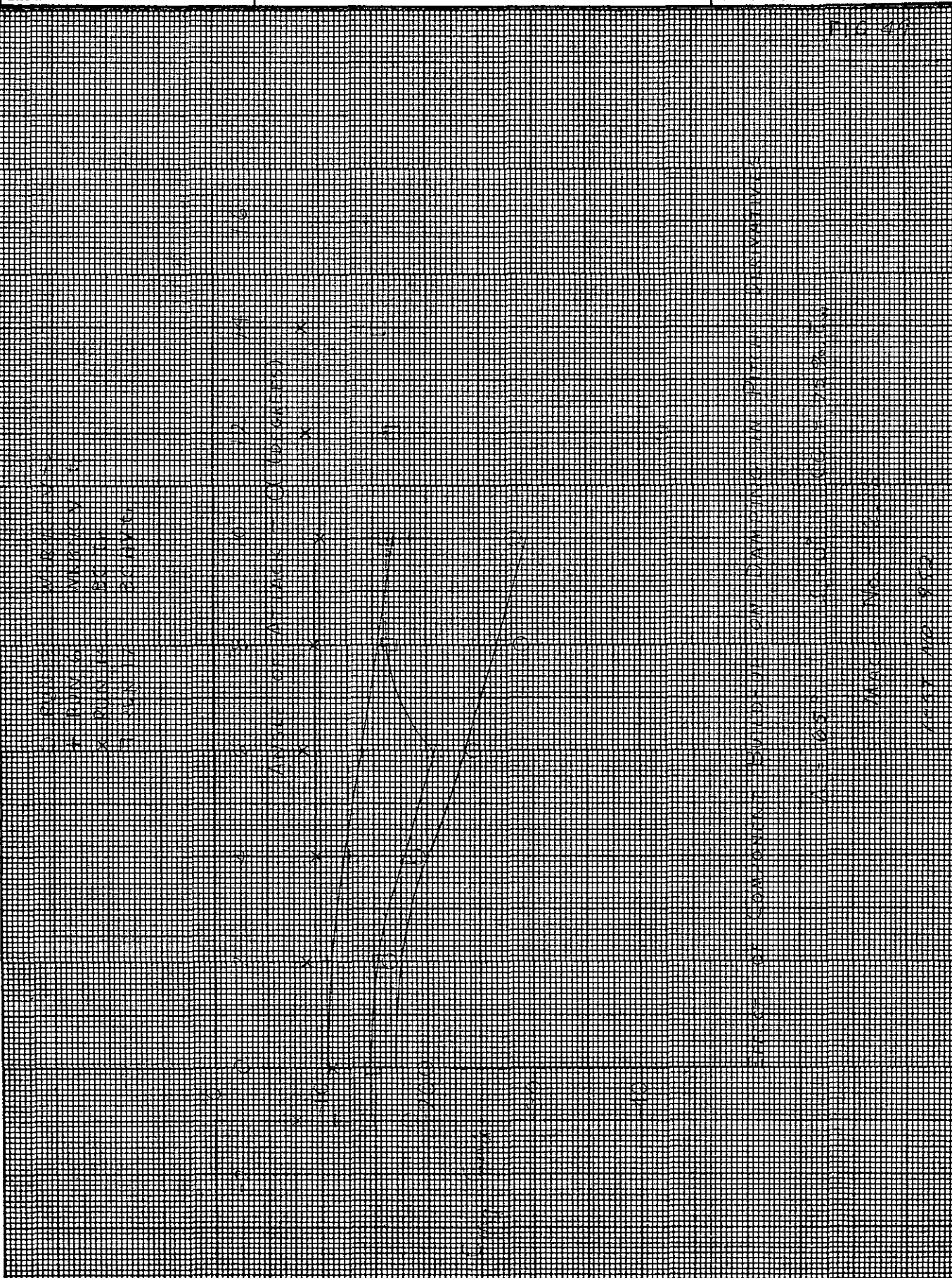
PAGE NO. 75 OF 145

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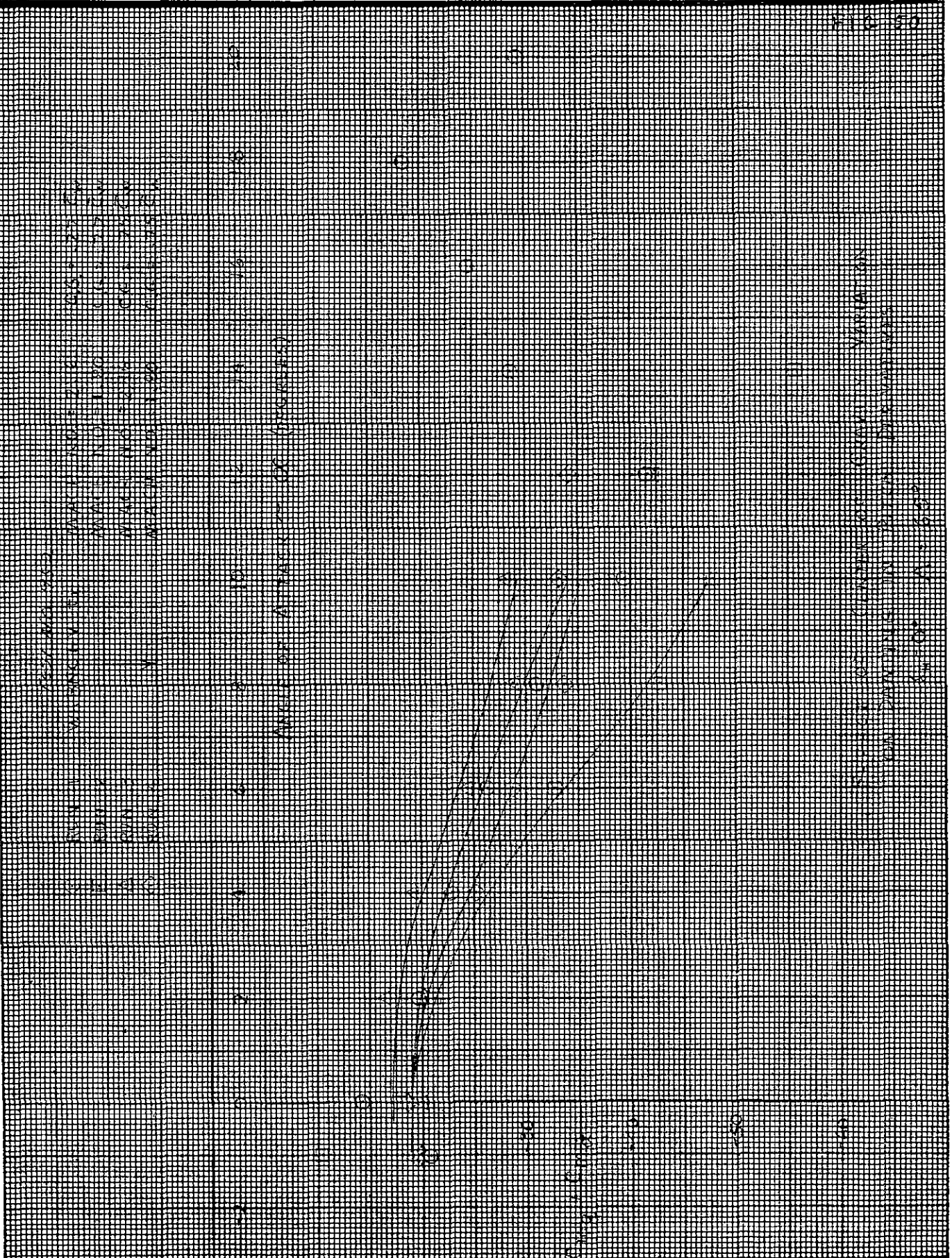


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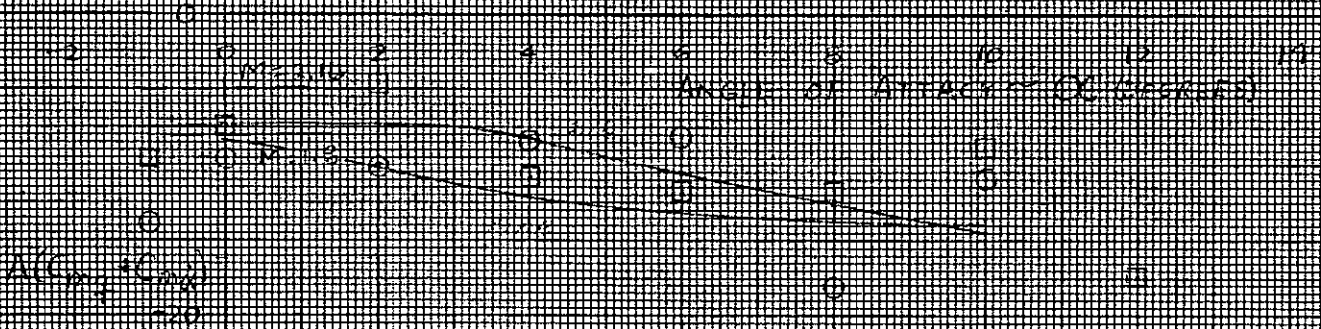
DATE:

MODEL NO.

FIG. 51

○ RUN 11 - RUN 13 MACH NO. = 1.0
 ○ RUN 0 - RUN 12 MACH NO. = 2.16
 — — — ESTIMATED HORIZONTAL INPUT

RUN	WING	WING	WING	WING	WING	WING	WING	WING
RUN 10	WING	WING	WING	WING	WING	WING	WING	WING
RUN 11	WING	WING	WING	WING	WING	WING	WING	WING
RUN 12	WING	WING	WING	WING	WING	WING	WING	WING
RUN 13	WING	WING	WING	WING	WING	WING	WING	WING



HORIZONTAL TAIL INPUT INTO DAMPING IN PITCH
 DERIVATIVES - WING-ON

TEST NO. 912

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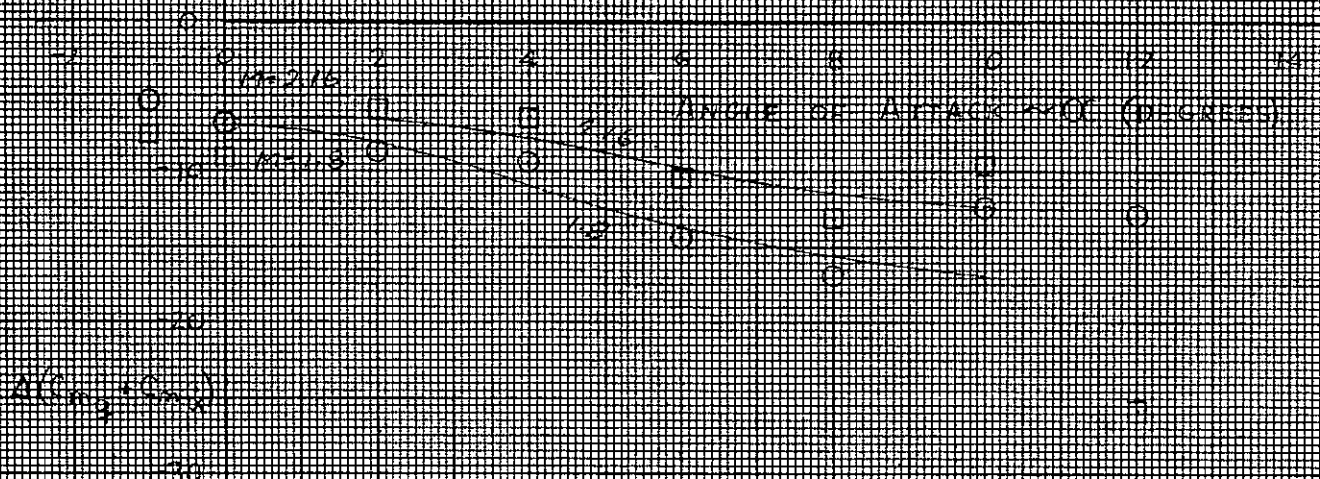
DATE:

MODEL NO.

FIG. 52

0 RUN 2 - RUN 7 TRACK NO. 210
 0 RUN 3 - RUN 6 TRACK NO. 210
 --- ESTIMATED HORIZONTAL INPUT

RUN	W	H	A	M	CG
RUN 2	0.7	0.5	0.6	0.5	0.5
RUN 3	0.7	0.5	0.6	0.5	0.5
RUN 6	0.7	0.5	0.6	0.5	0.5
RUN 7	0.7	0.5	0.6	0.5	0.5



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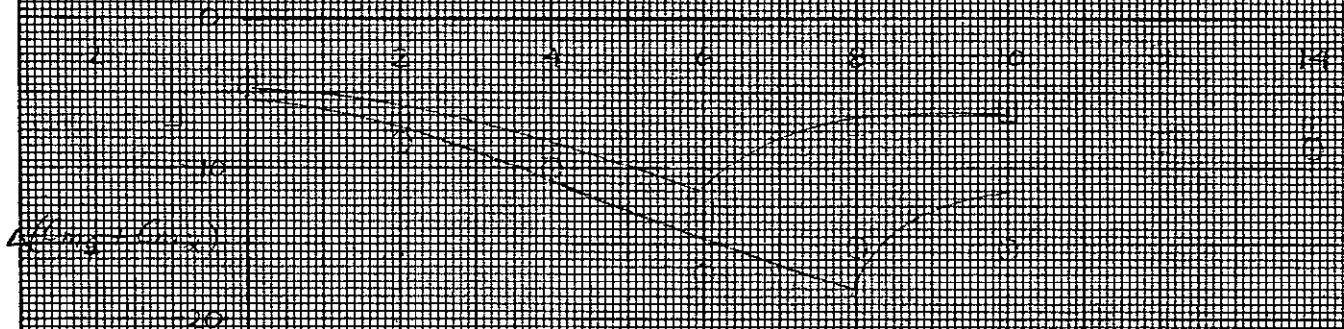
MODEL NO.

FIG 53

C RUN 18 - RUN 6 MACH NO = 1.8
 S RUN 7 - RUN 14 MACH NO = 2.16


	δ_{eff}	M	CC
RUN 14	BC + HV	2.16	25.7% SW
RUN 16	BC + HV	1.80	
RUN 17	BC + HV	2.16	
RUN 18	BC + HV	1.80	N

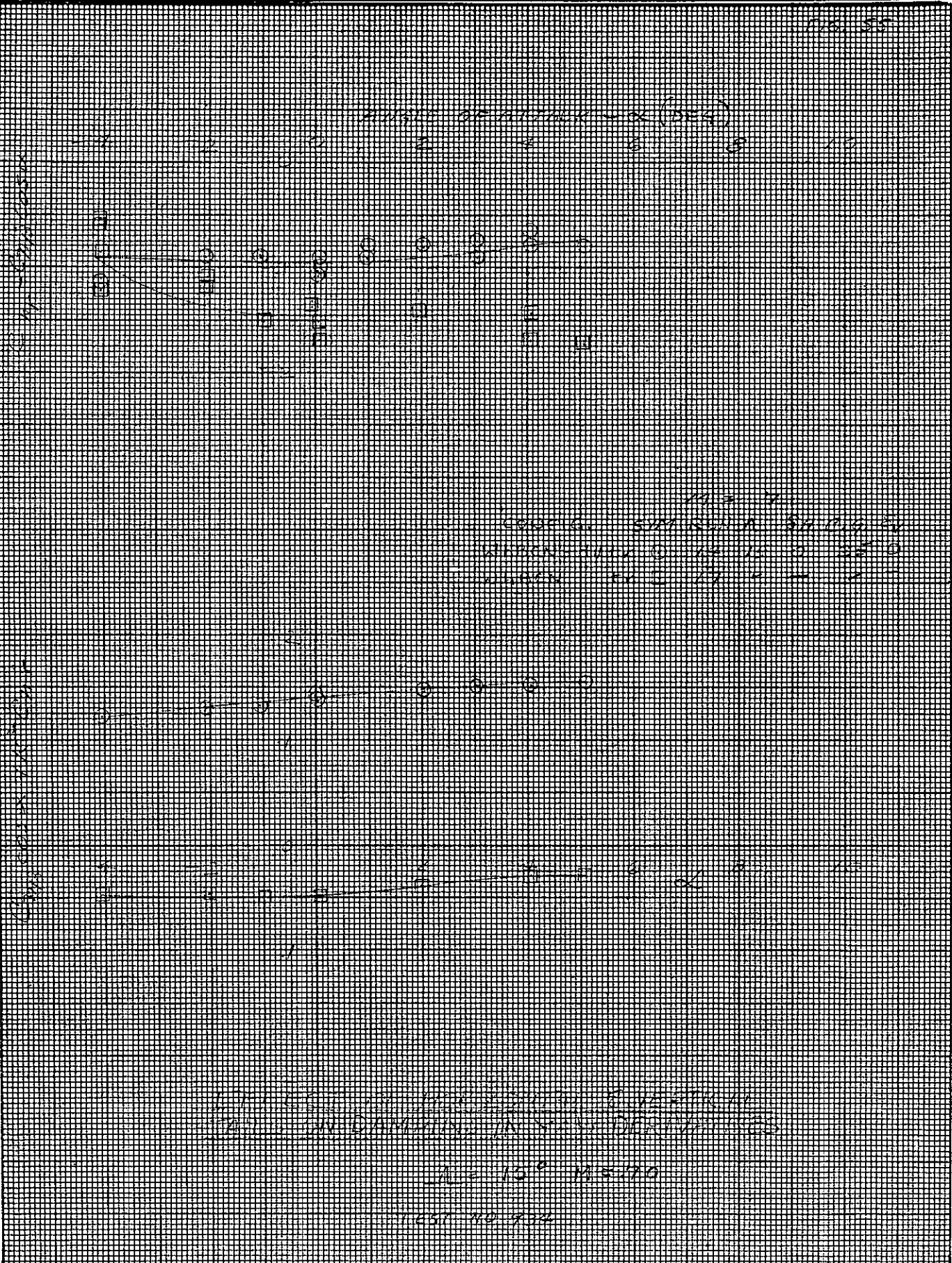
ANGLE OF ATTACK - CC (DEGREES)



HORIZONTAL TAIL INPUT W/O DAMPING IN PITCH DERIVATIVES
- WING OFF

TEST NO. 953

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DATE:		MODEL NO.





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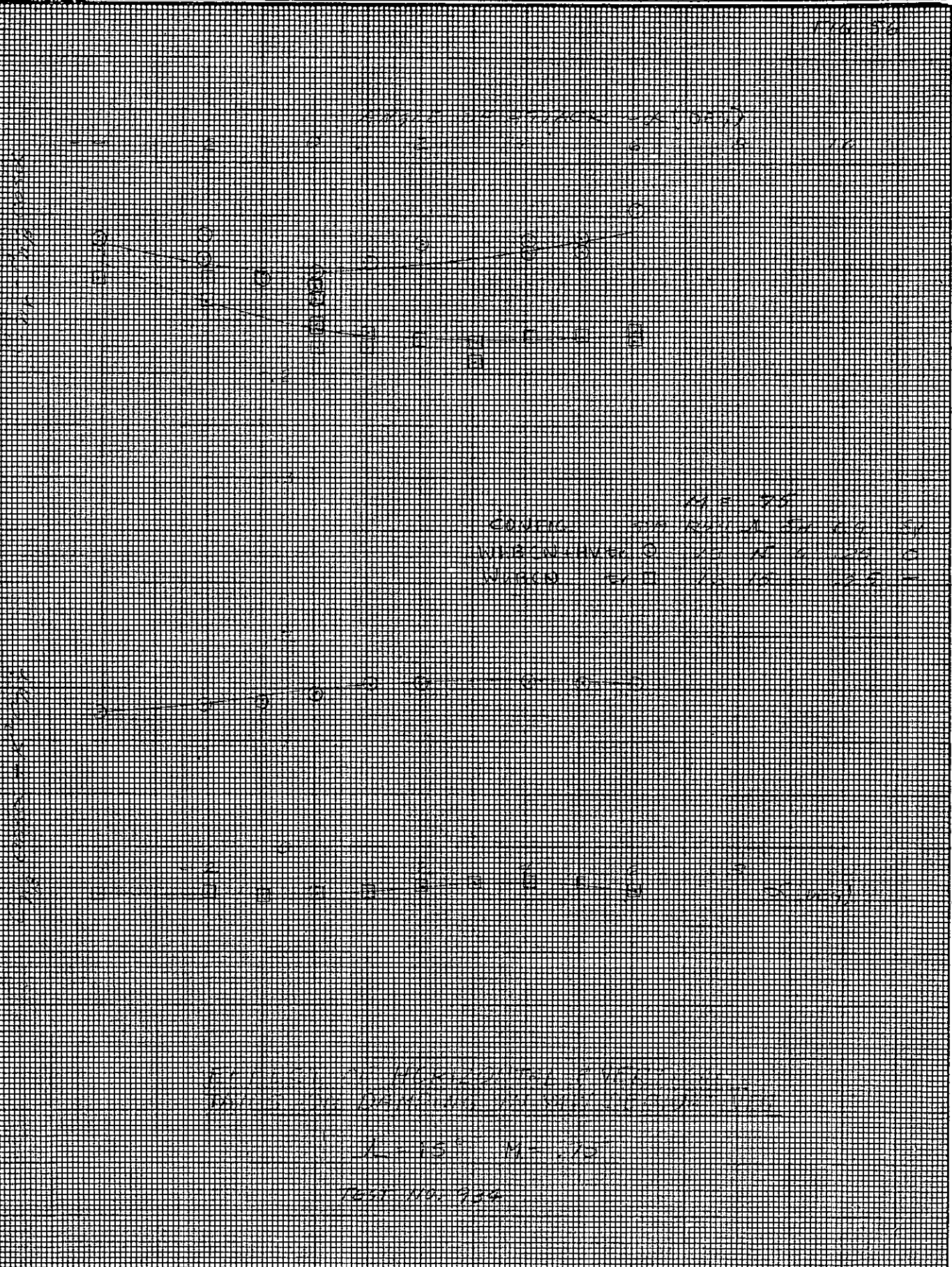
PAGE NO. 82 OF 145

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DATE:

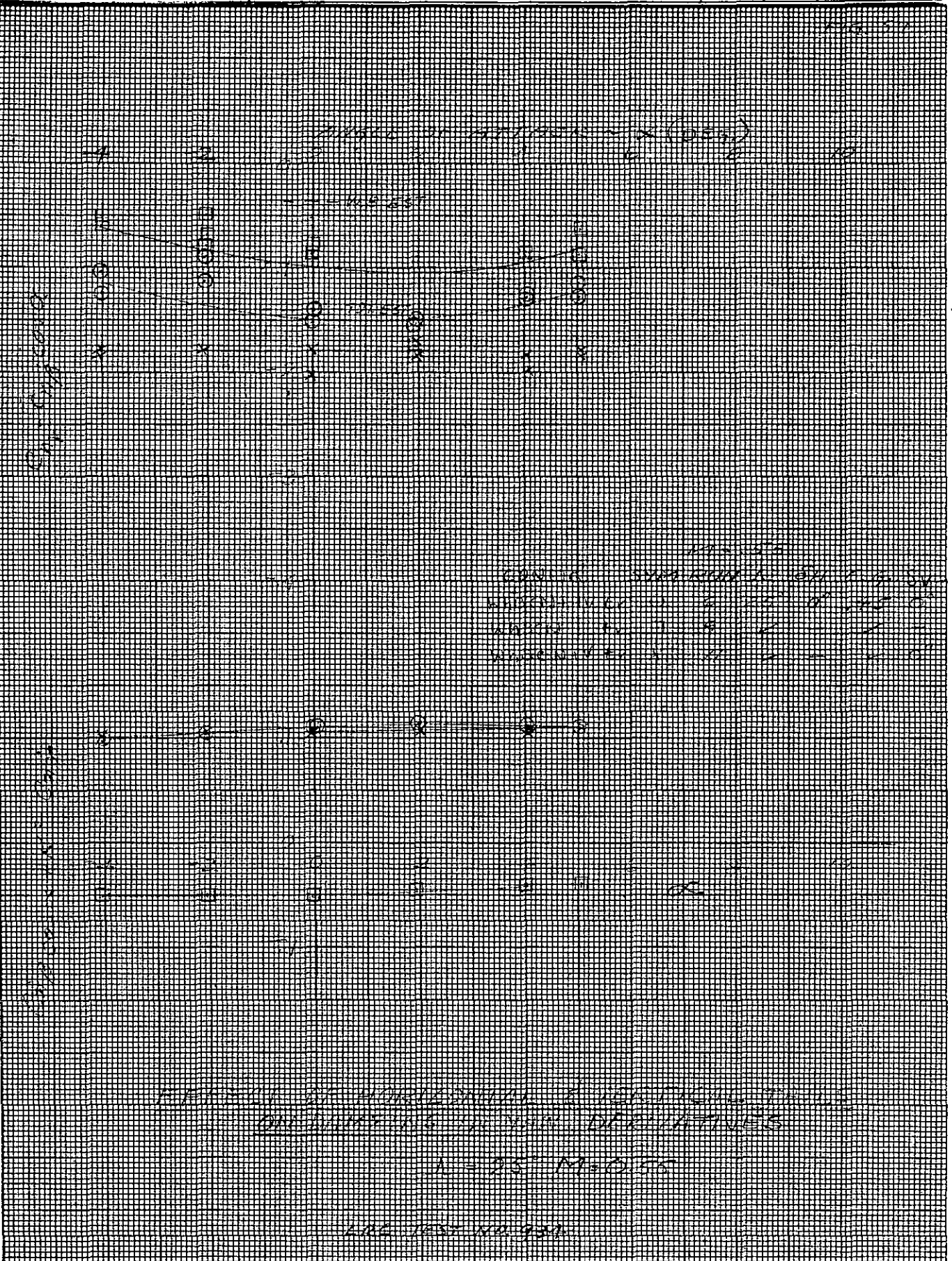


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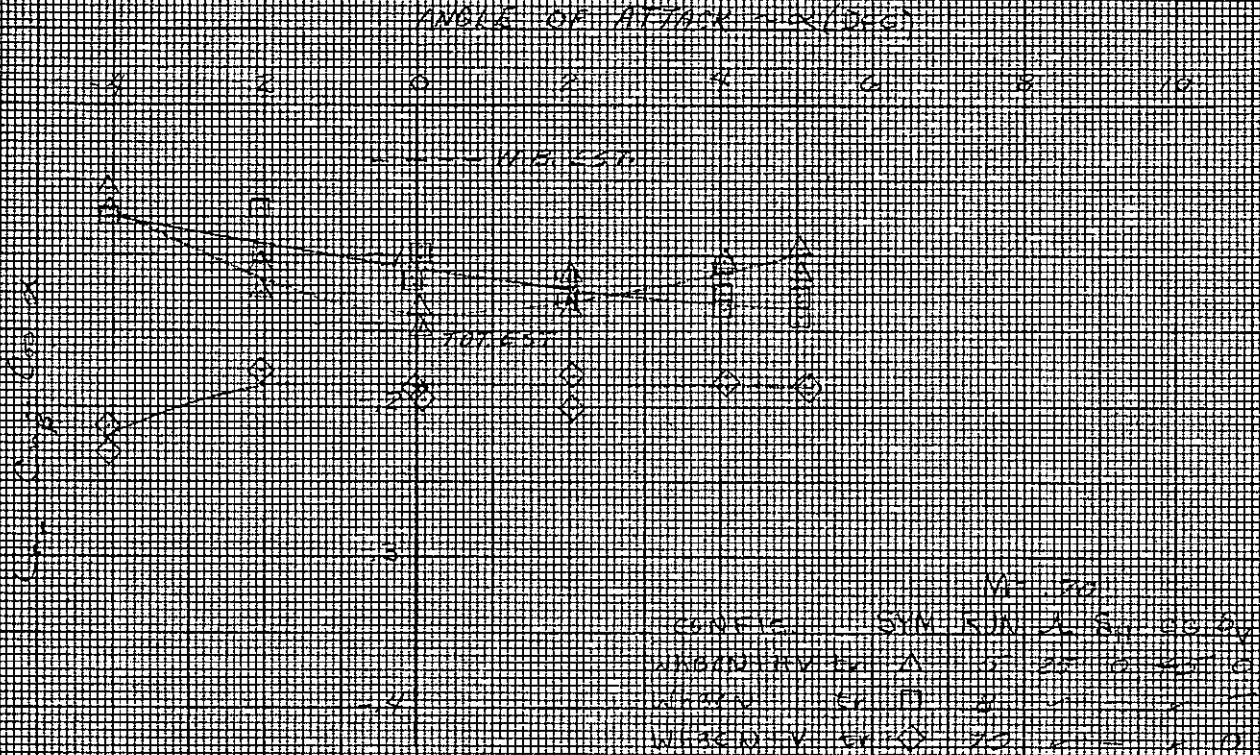
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FIG. 58



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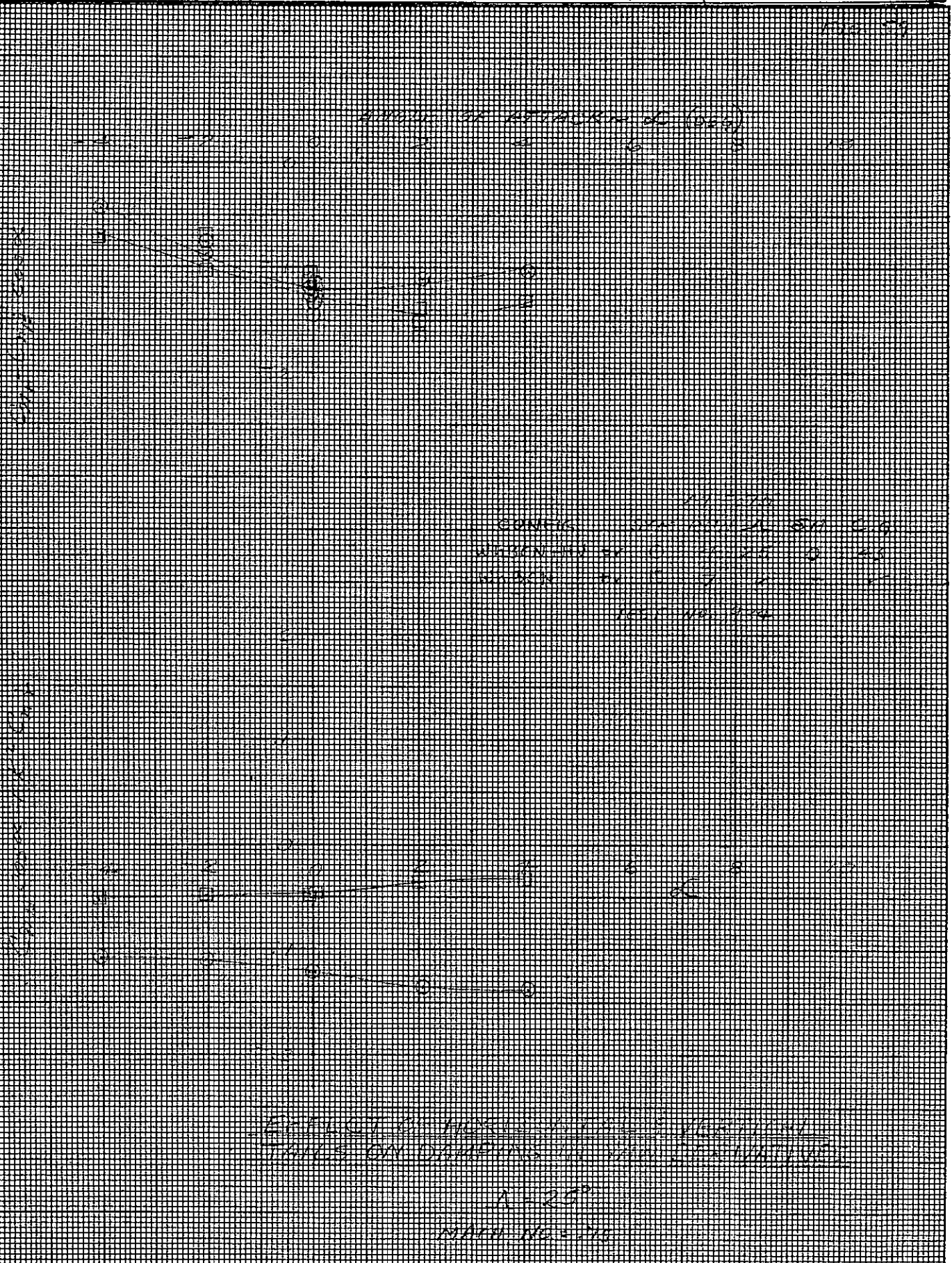


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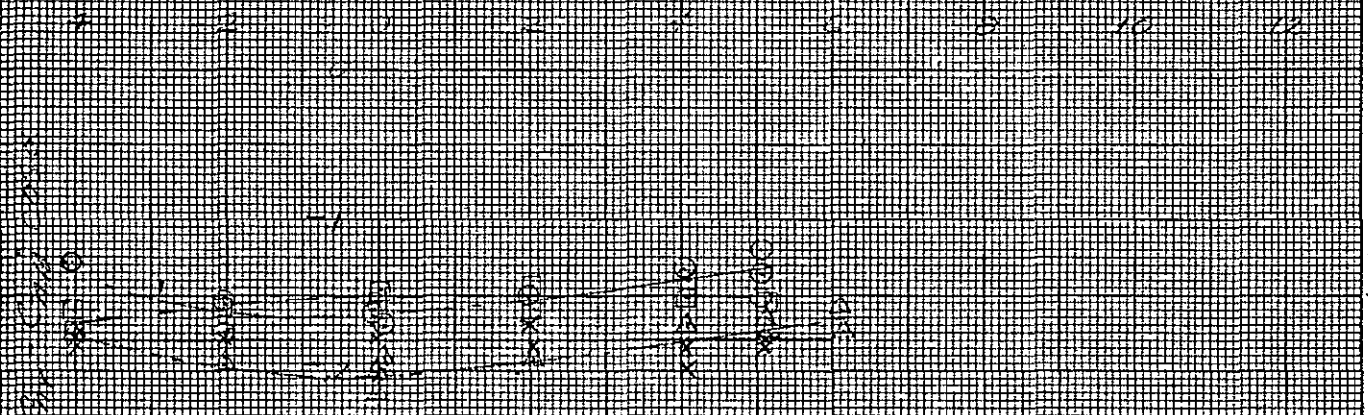
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REPORT NO NA-72-82

DATE:

MODEL NO.

ANGLE OF ATTACK SIDE (DEG)



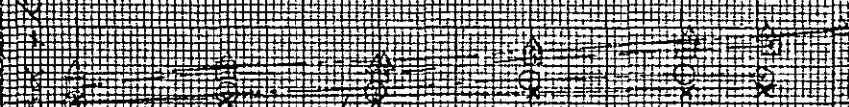
NOTE

BY BODY WITH FILLER BLOCKS BETWEEN

W/OCU + VIB + FUSELAGE

M=0.50

CONFIG	SYNTHETIC	2.5	5.0	10.0	15.0
W/OCU + VIB + A	72	25	0	25	65
W/OCU + VIB + D	74	✓	✓	✓	65
W/OCU + VIB + X	71	✓	✓	✓	65
W/OCU + VIB + O	65	✓	0	✓	55



ANGLE OF ATTACK SIDE (DEG)

EFFECT OF FILLER BLOCKS ON

DAMPING IN FOUR DERIVATIVES

M=0.50

TEST NO 930

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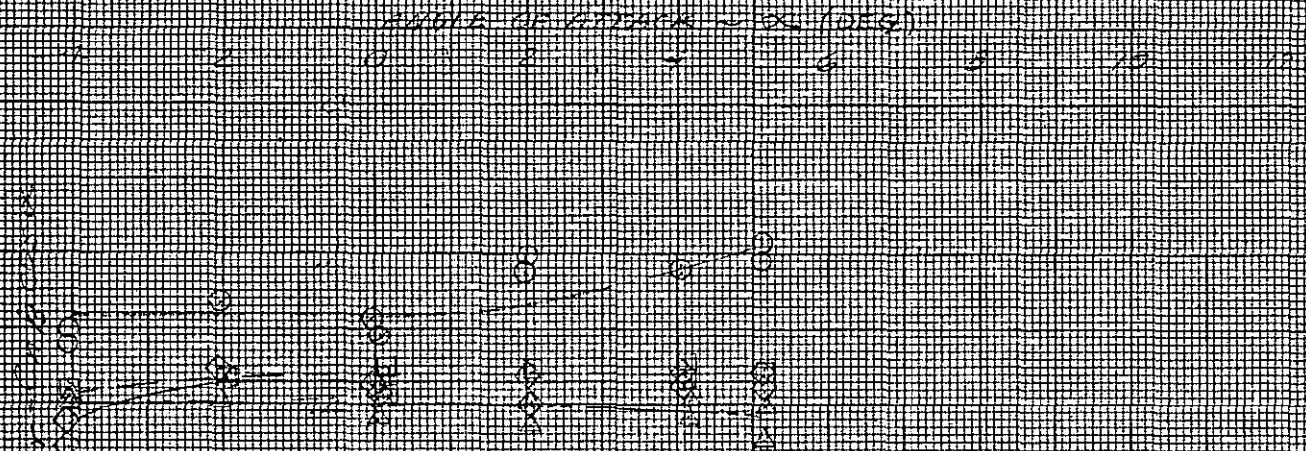
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DATE:

MODEL NO.

FIG. 61



NOTE:

NO BODY WITH FILLER BLOCKS BETWEEN
HOLD TAIL & FUSELAGE

M=7

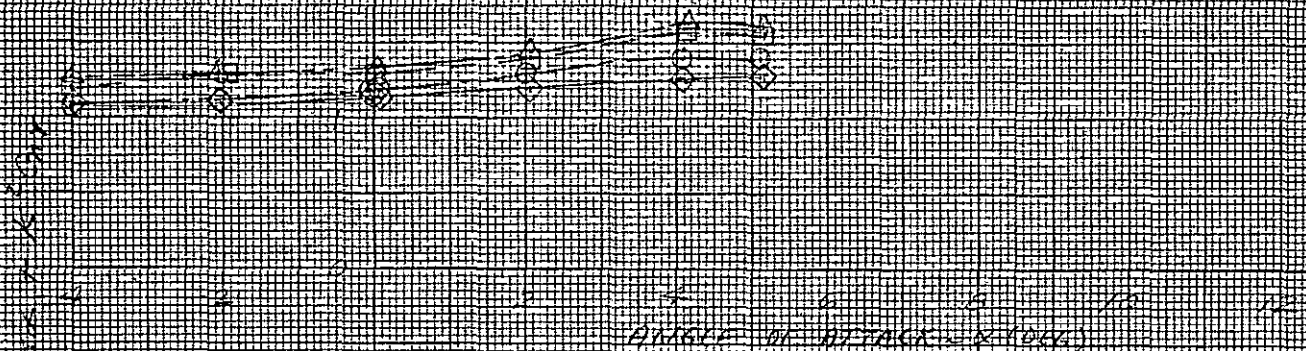
FUSelage - SW RIN A 20.16

WINGEN - VIV A 73.25 0.745

WINGEN - VIV B 72.2 0.74

WINGEN - VIV C 39.0 0.7

WINGEN - VIV D 70.0 0.7



EFFECT OF FIN DEFLECTION

DURING IN-TAKE DECELERATION

M=7.0

TEST NO. 939



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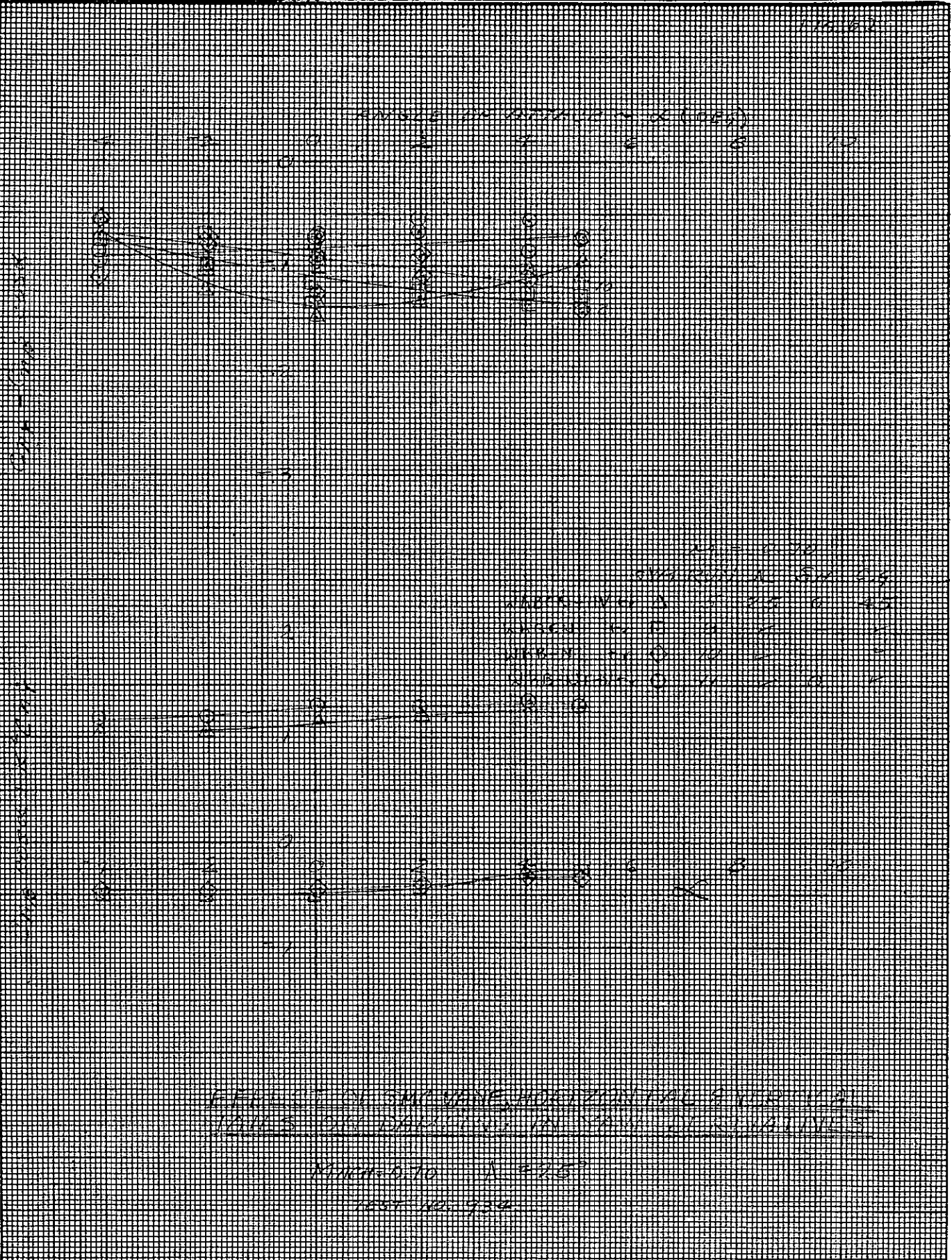
PAGE NO 88 OF 145


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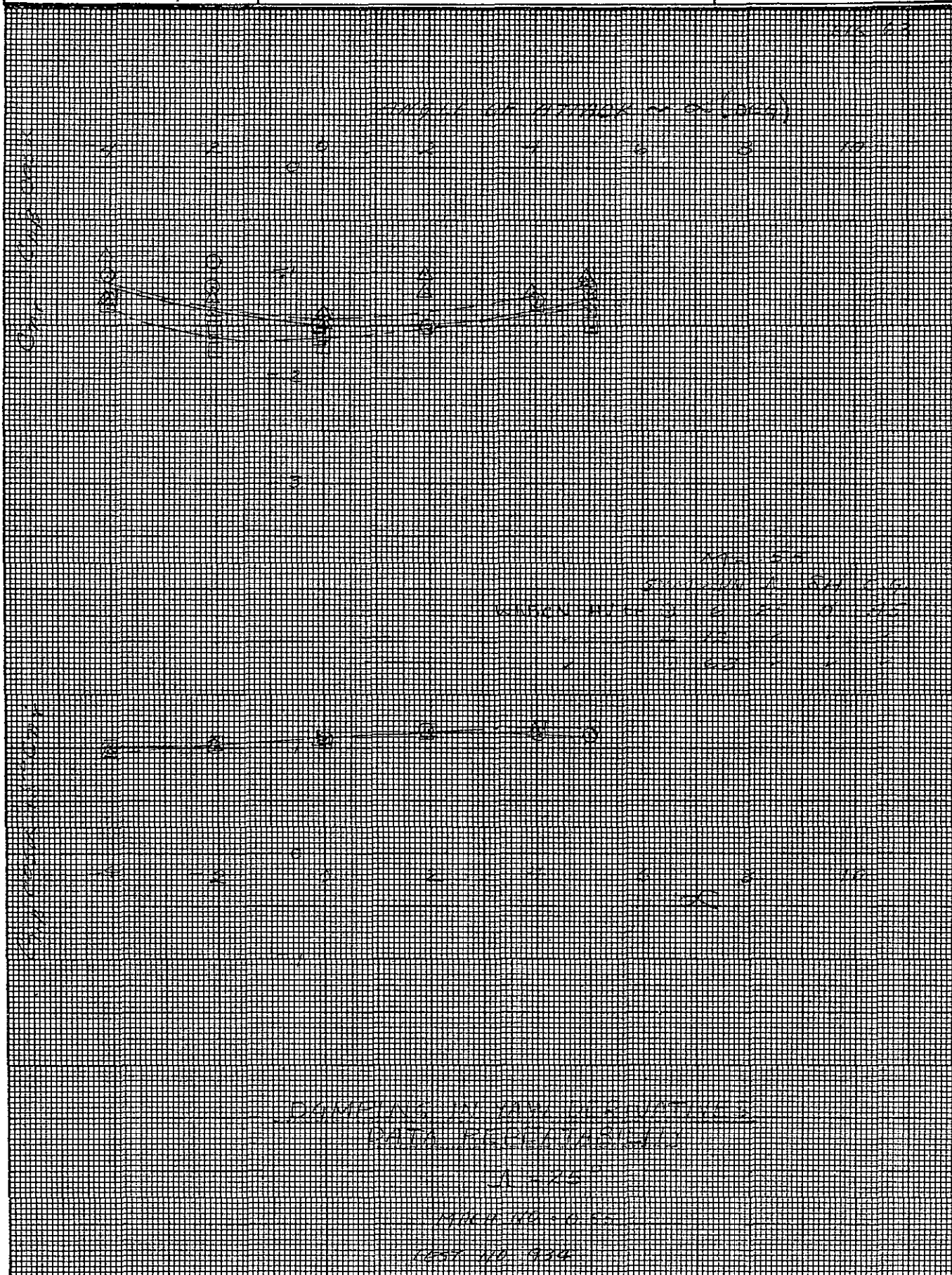
REPORT NO NA-72-82

DATE:


MODEL NO.

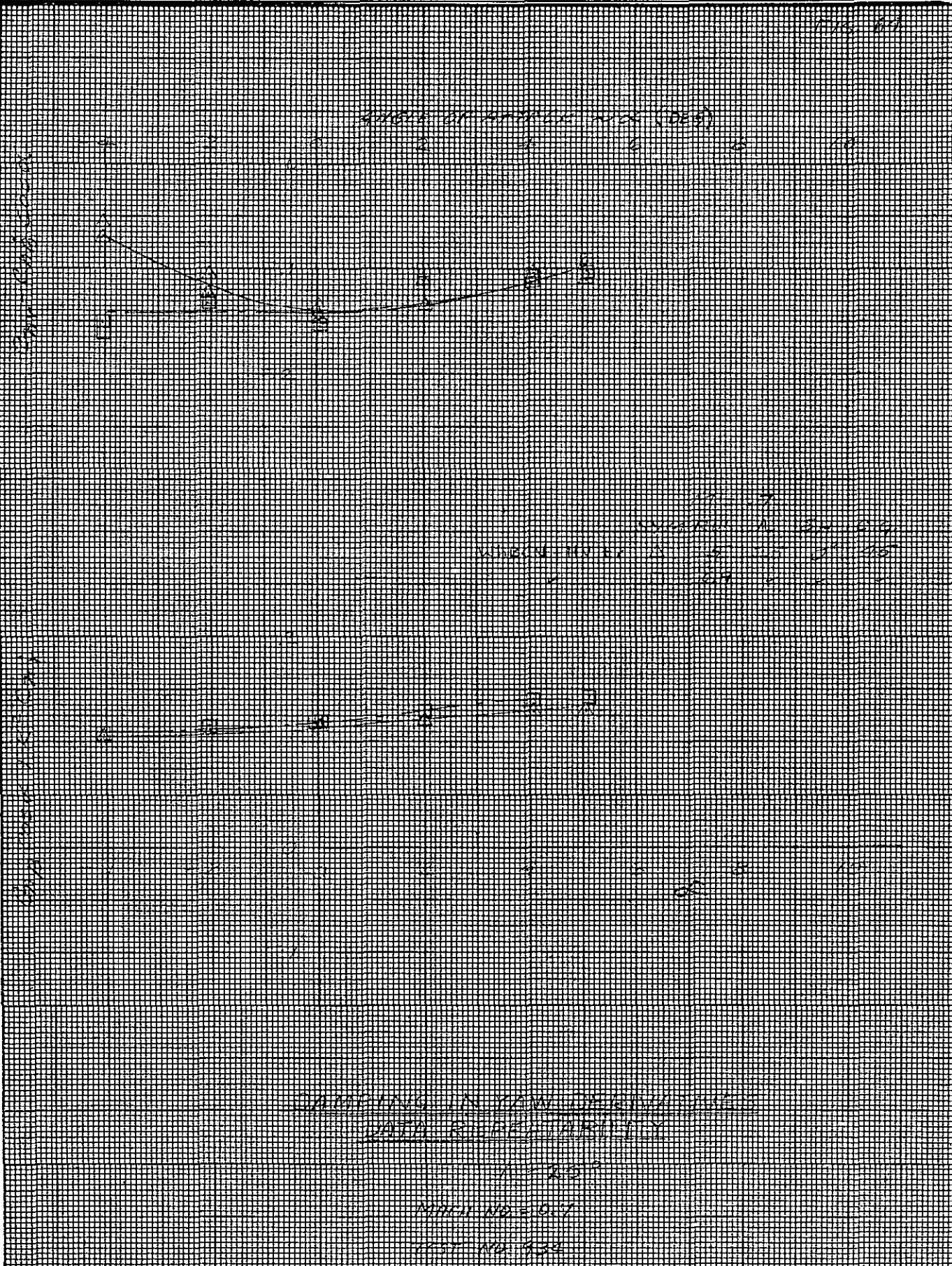


FORM 1021B REV 10-70		
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DATE:		MODEL NO.



✓
✓

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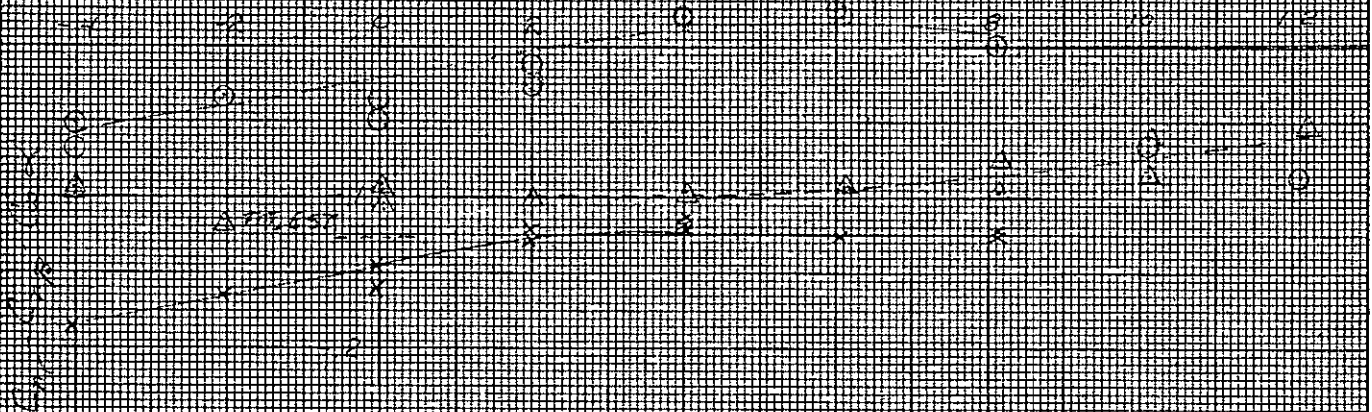
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REPORT NO. NA-72-82

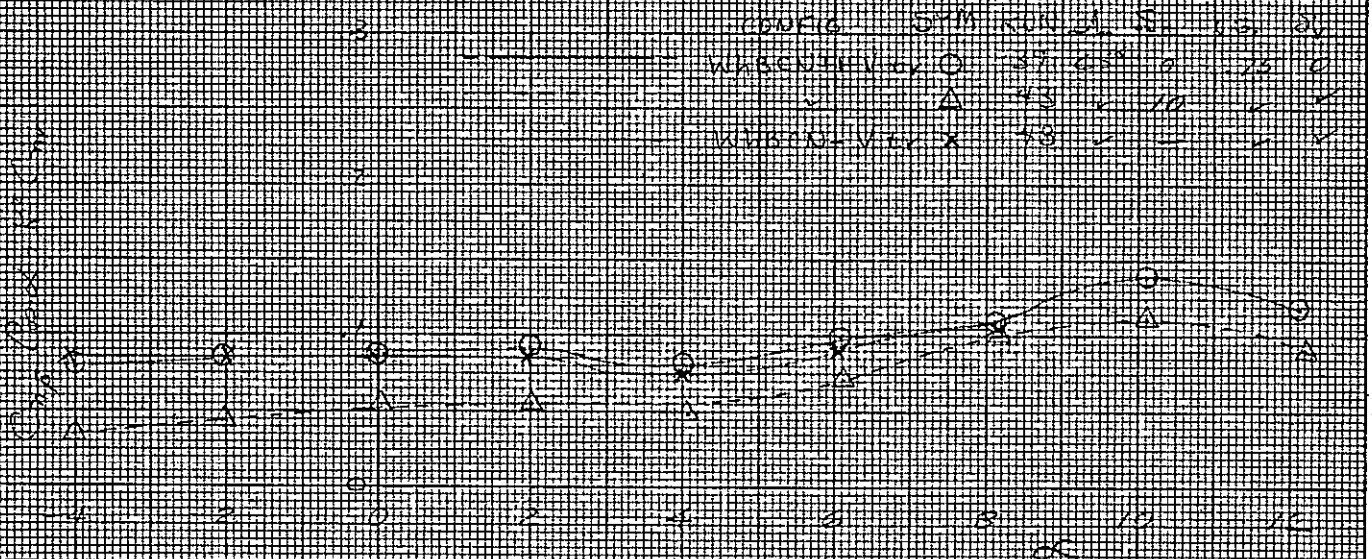
DATE:

MODEL NO

FIG. 66

ANGLE OF ATTACK $\sim \alpha$ (DEG)

M = 75

EFFECT OF HORIZONTAL & VERTICAL TAILS
ON DAMPING IN YAW DERIVATIVES

M = 65 Mach = 0.75

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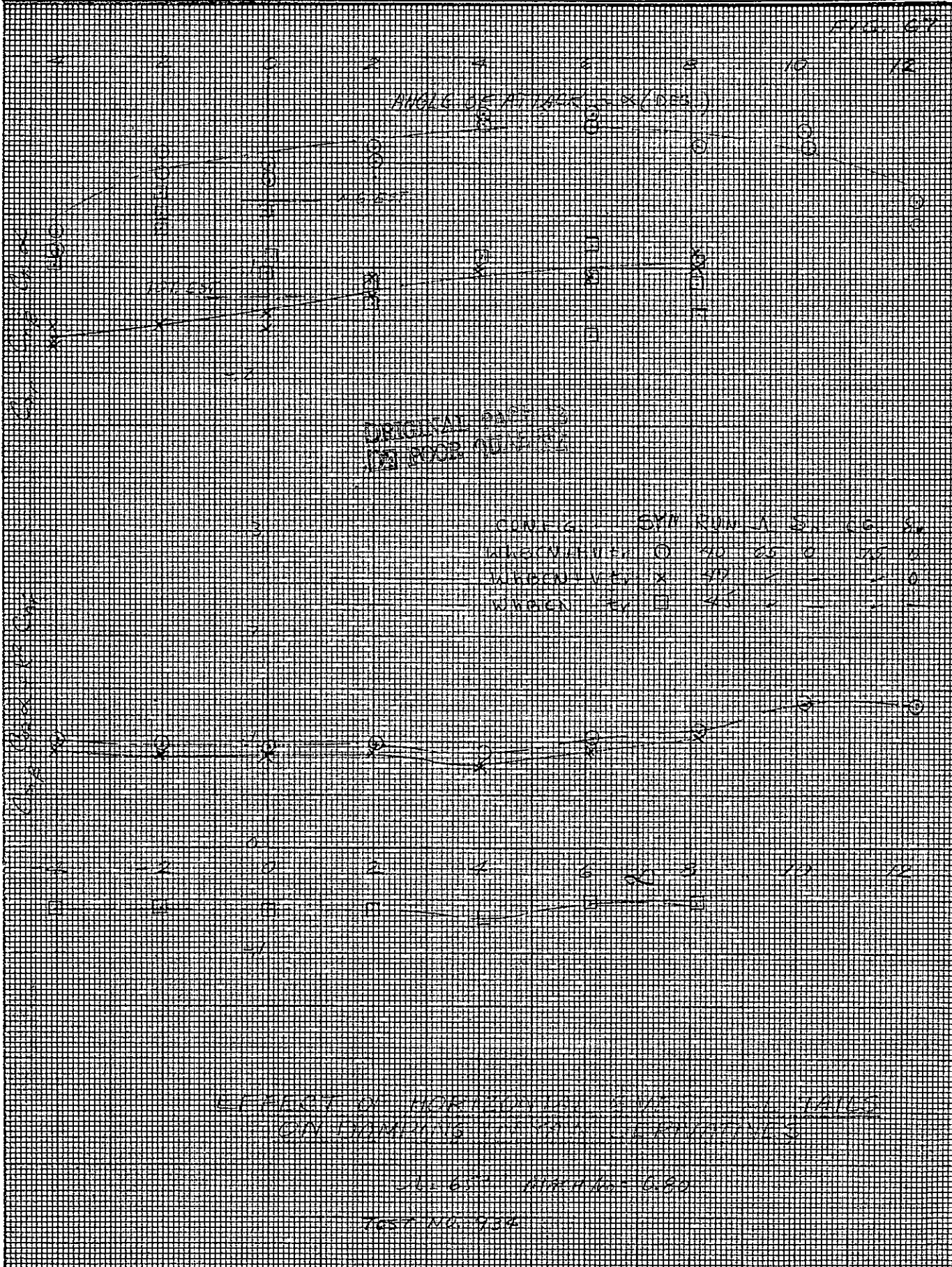


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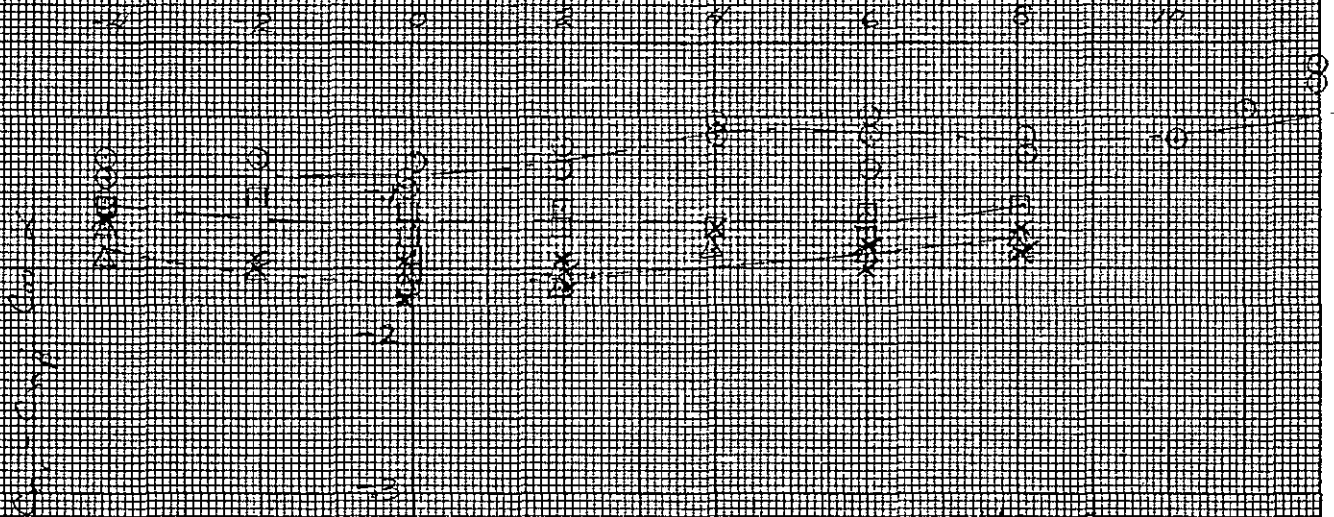
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DATE:

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FIG 68

ANGLE OF ATTACK α (DEG)

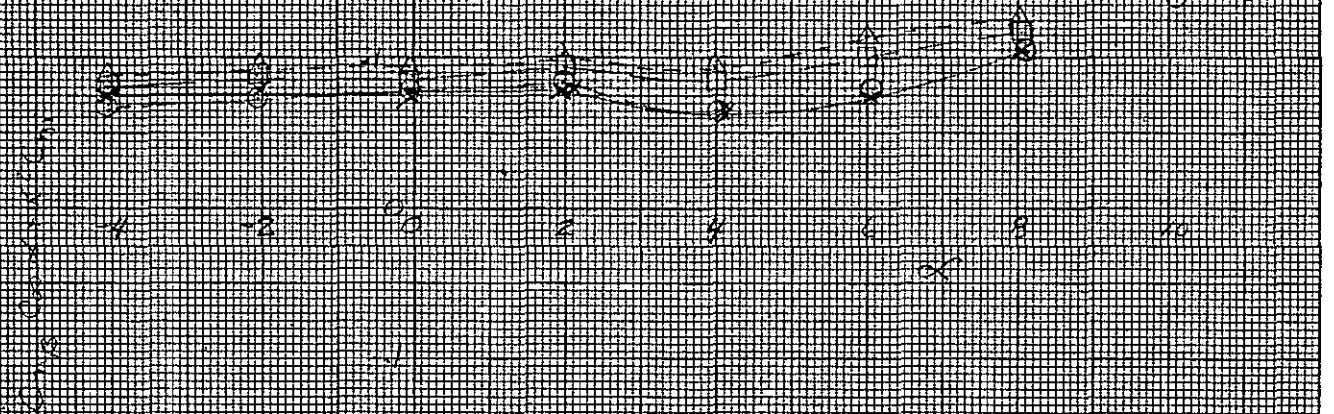
DATE:

TIME:

SYM RUN J. S. 00 S.

80 BODY WITH FLARE PLANKS
BEFORE HOISTING OFF BASE

CL	CL	CL	CL	CL	CL	CL	CL	CL	CL
0.2	0.4	0.6	0.8	1.0	1.0	1.0	1.0	1.0	1.0
0.2	0.4	0.6	0.8	1.0	1.0	1.0	1.0	1.0	1.0
0.2	0.4	0.6	0.8	1.0	1.0	1.0	1.0	1.0	1.0
0.2	0.4	0.6	0.8	1.0	1.0	1.0	1.0	1.0	1.0
0.2	0.4	0.6	0.8	1.0	1.0	1.0	1.0	1.0	1.0

EFFECT OF ELLIPTIC BUCKS ON
CLAMPING IN VAIN DERIVATIVES

M-088 2A-00

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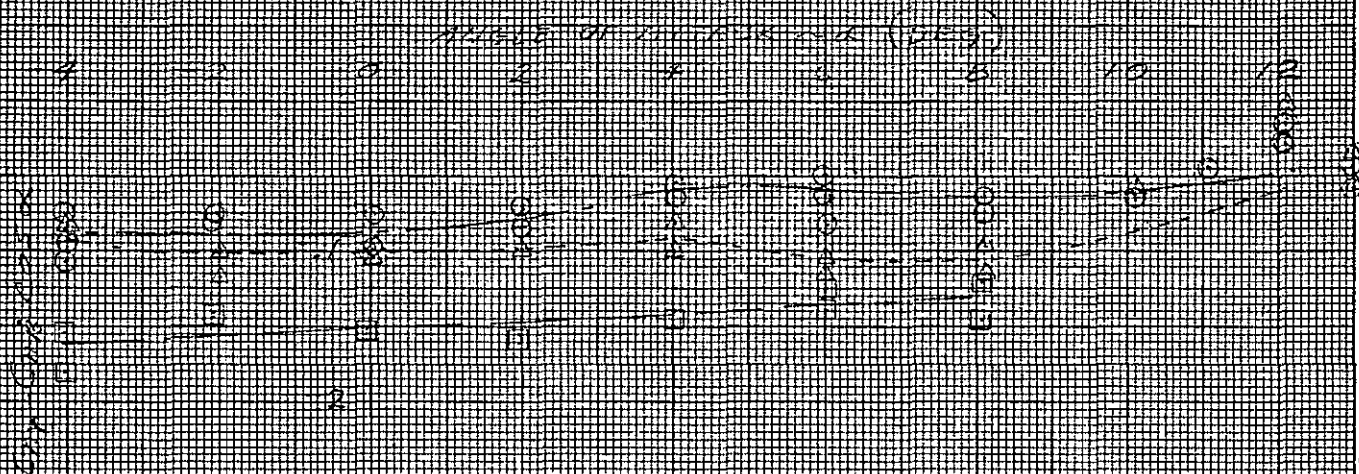
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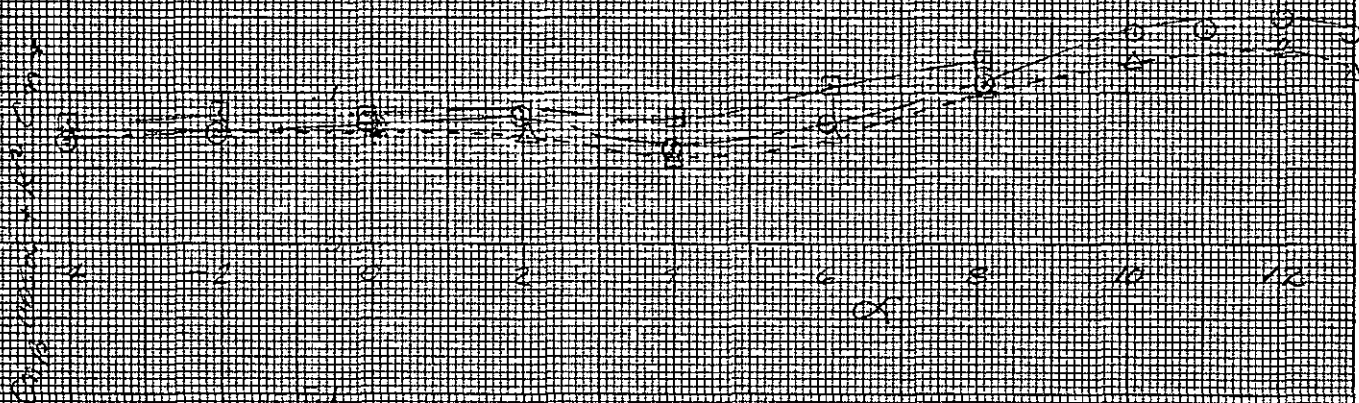
DATE:

MODEL NO.

115 10

ORIGINAL PAGE
OF POOR QUALITY

CONFIG. 11-5T
 APPROX. TORSION 61.63 x 10⁻³
 APPROX. TORSION 42.5 x 10⁻³
 APPROX. TORSION 31.7 x 10⁻³



ANGLE OF TWIST (DEG) vs. TORQUE (in-lb)
 ON DURING IN THE DURING

1-65° M. 0.58

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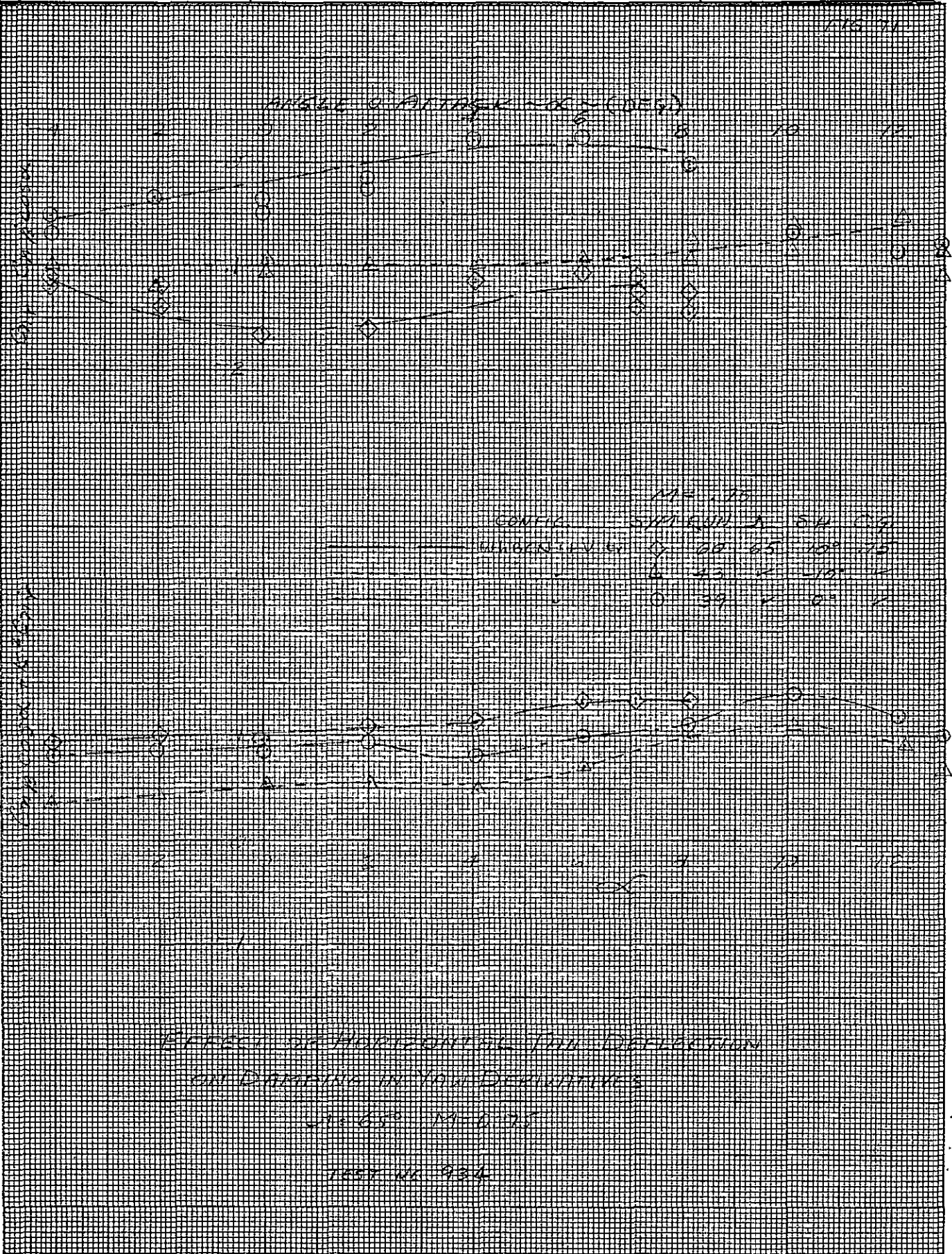
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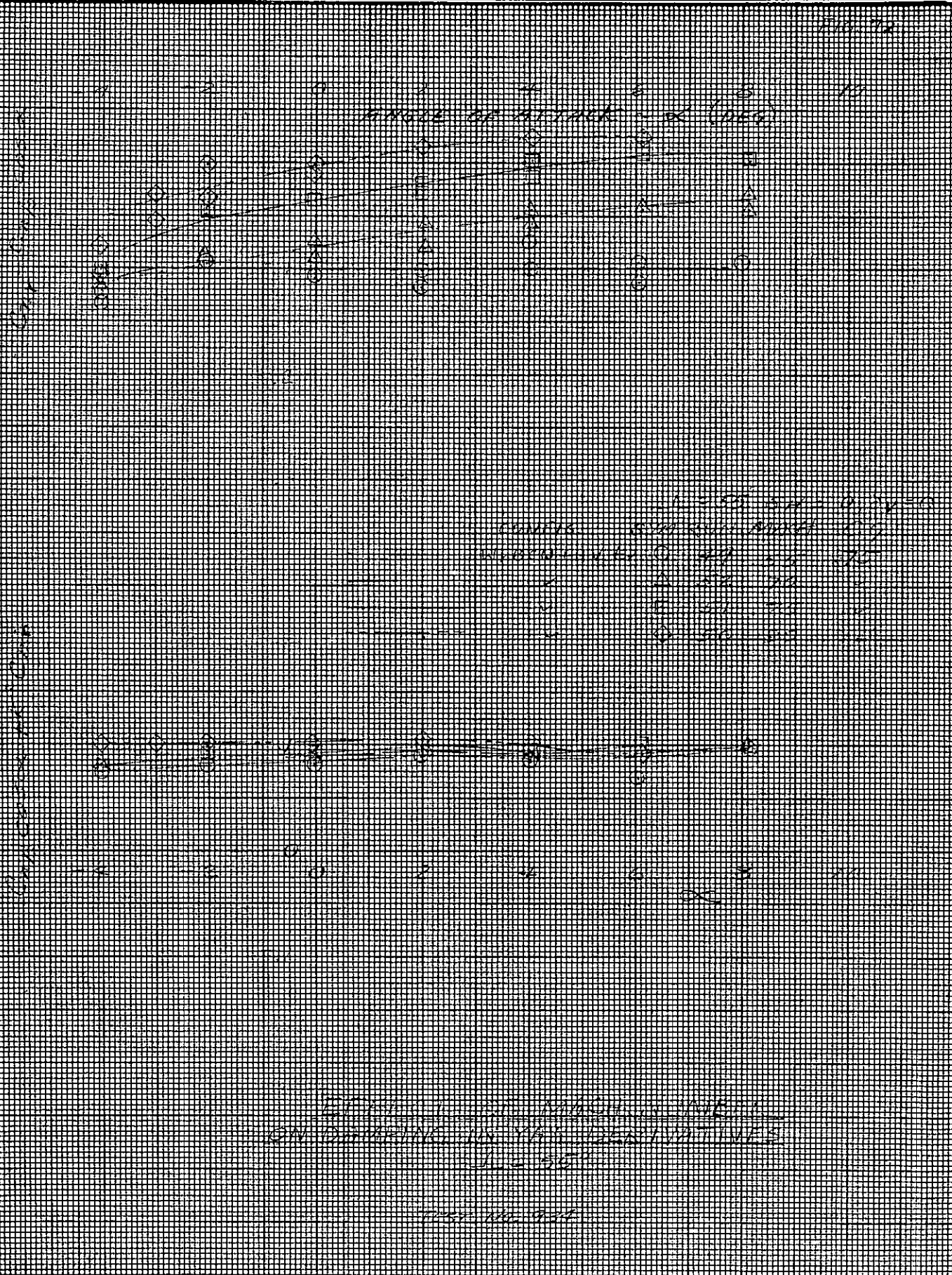


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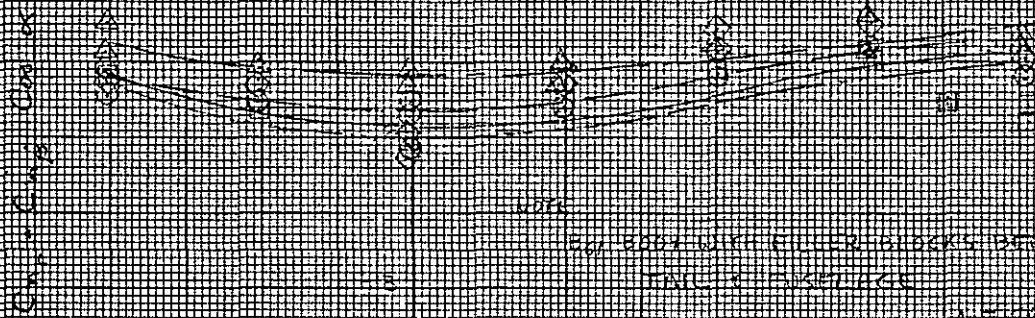
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FIG 76

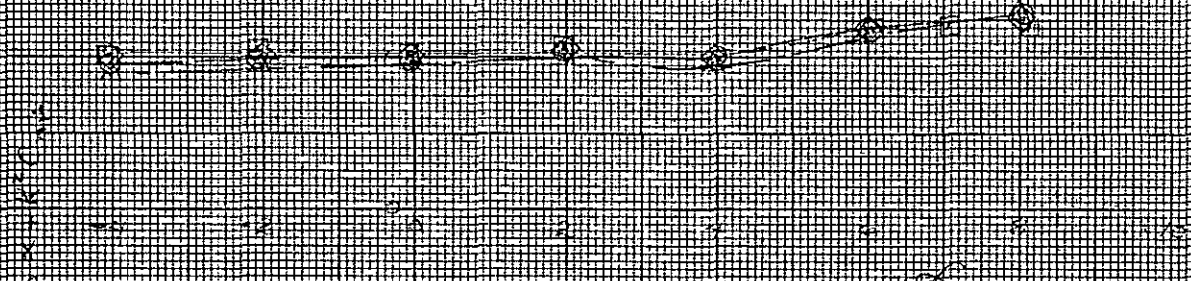
ANGLE OF ATTACK - α (DEG)

-4 -2 0 2 4 6 8 10



FOR 2001 WING-FLYER BLOCKS BETWEEN SUBSONIC
TAIL & FUSELAGE

$\lambda = 45^\circ$	$31/30^\circ$	50°	60°
CONFIC	SYM	RT	LM
W/860 IN V C	02	30	30
	0	43	10
	0	07	17
	0	25	45



EFFECT OF MACH NUMBER
ON DAMPING IN YAW

$\lambda = 45^\circ$

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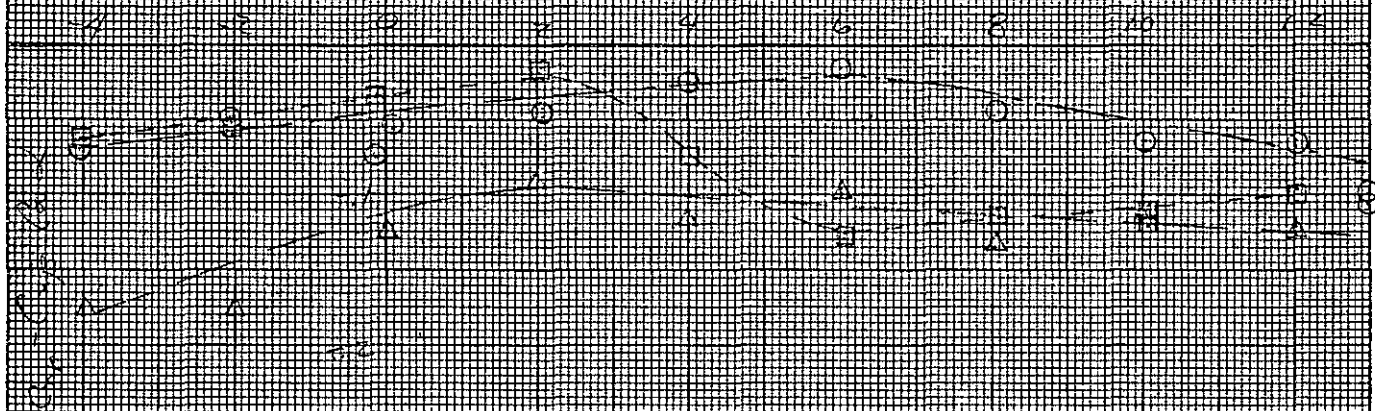
CHECKED BY:

REPORT NO NA-72-82

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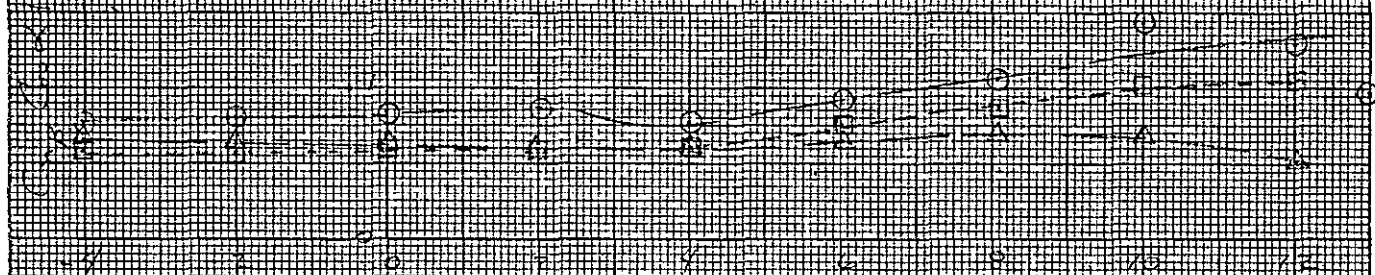
MODEL NO.

FIG 10

ANGLE OF ATTACK α (DEG.)

M 7

CONFIG	SPARK	WAGNER	CS	BY
BEHV BY	Δ 21	—	—	75 0
BEHV BY	Δ 21	—	—	10 1
WAGNER BY	○ 33	65	0	10 1

COMPOUND BUILDUP
WAGNER

MACH 0.20-0.70

TEST NO 730



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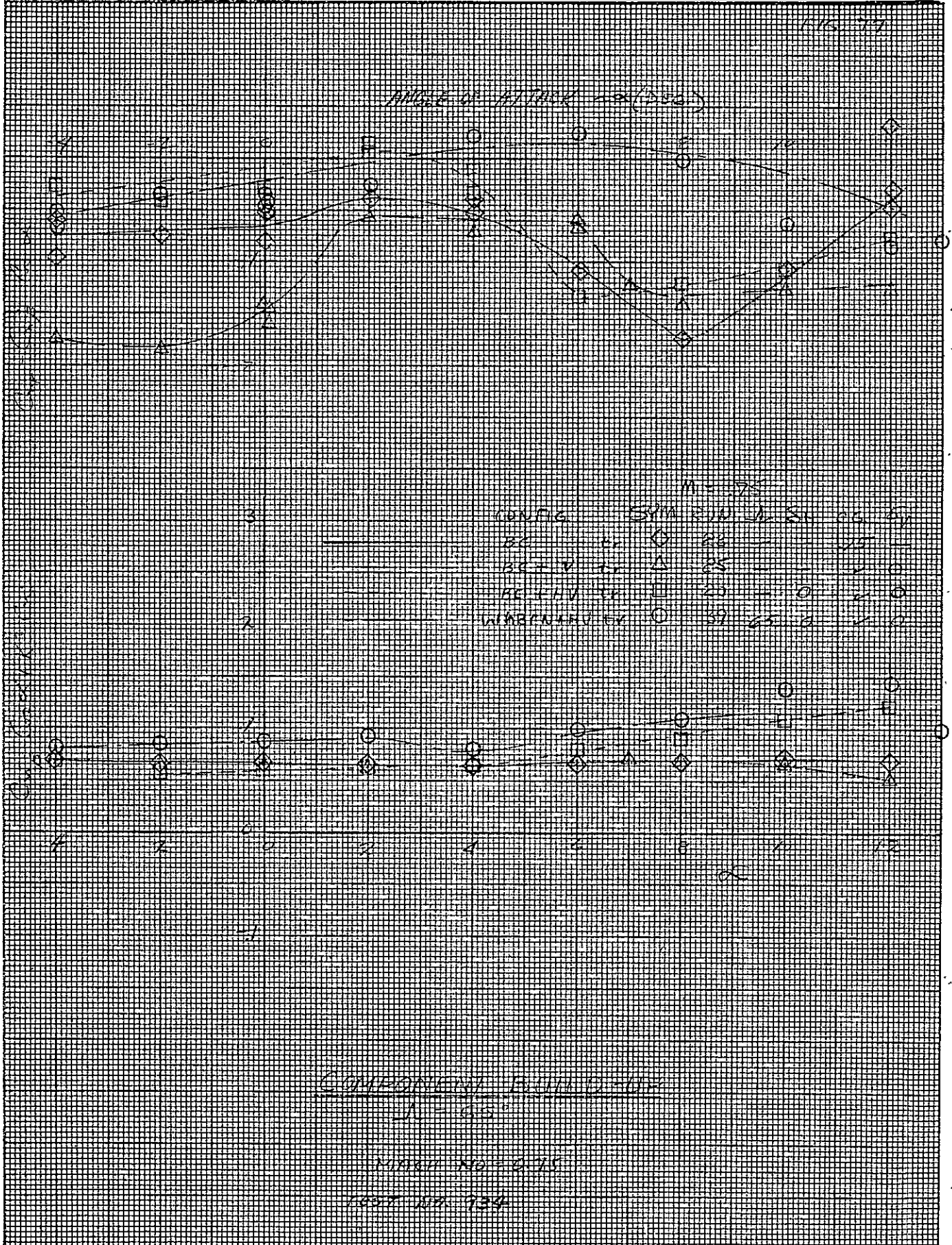
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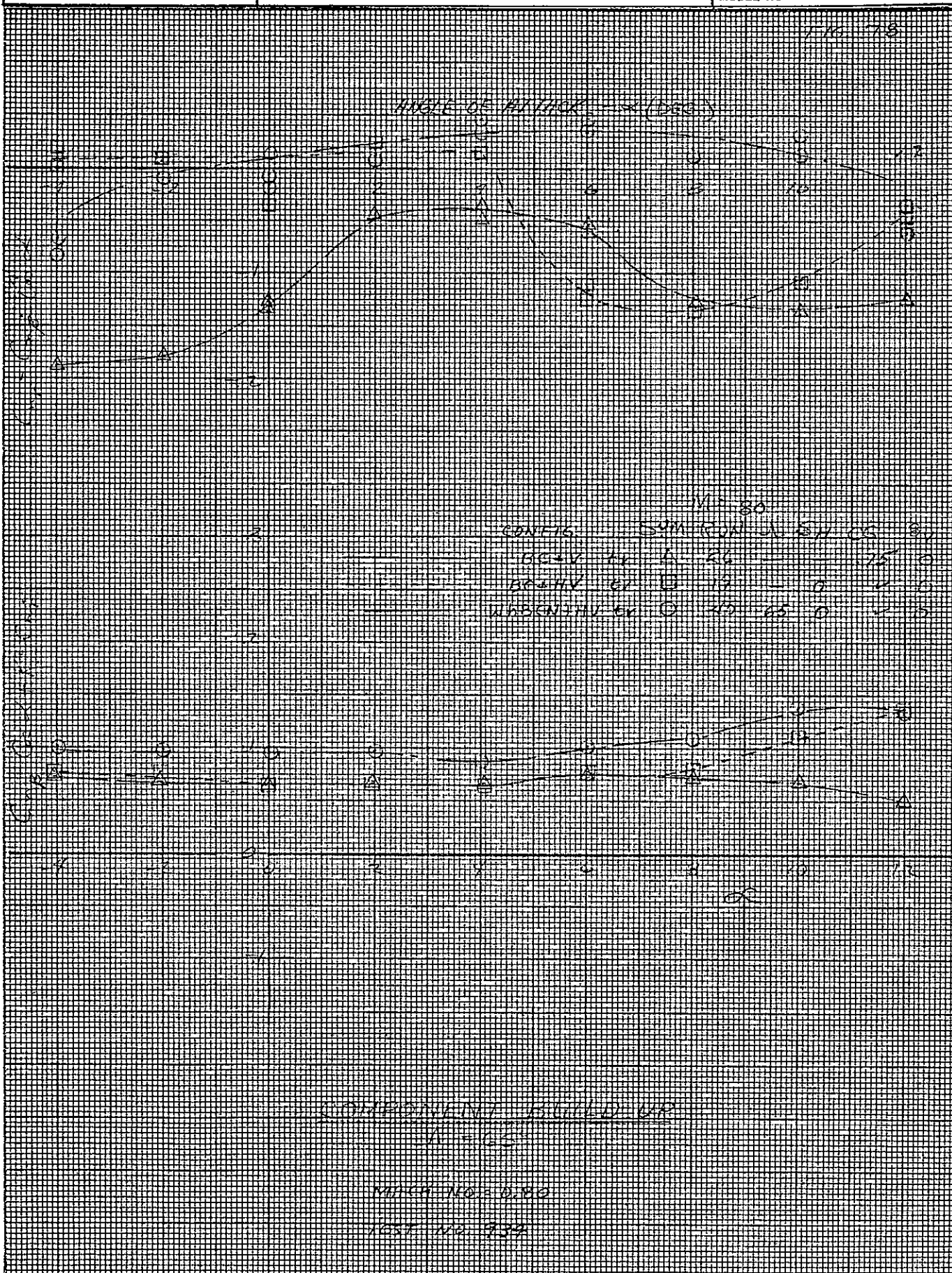
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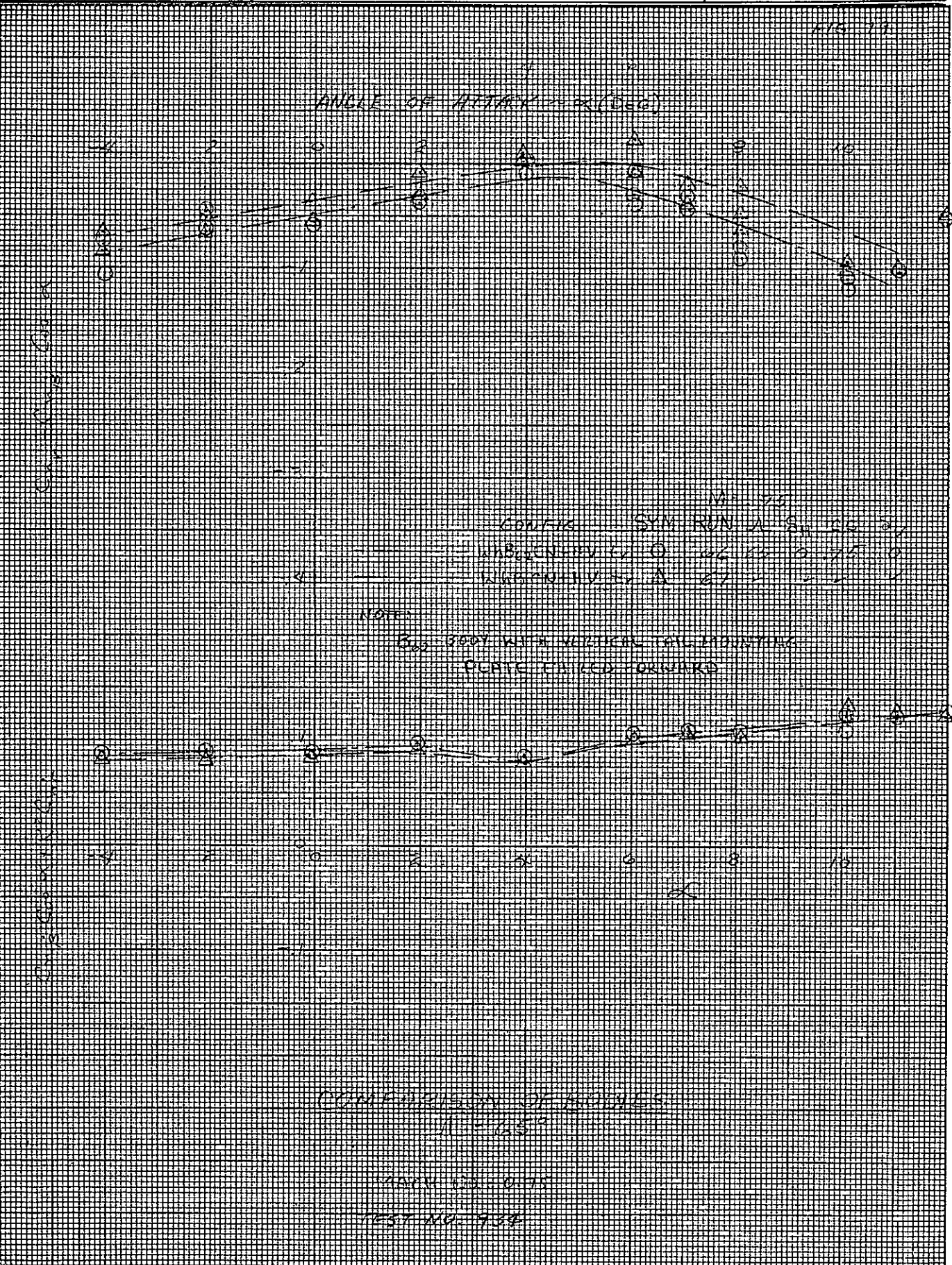
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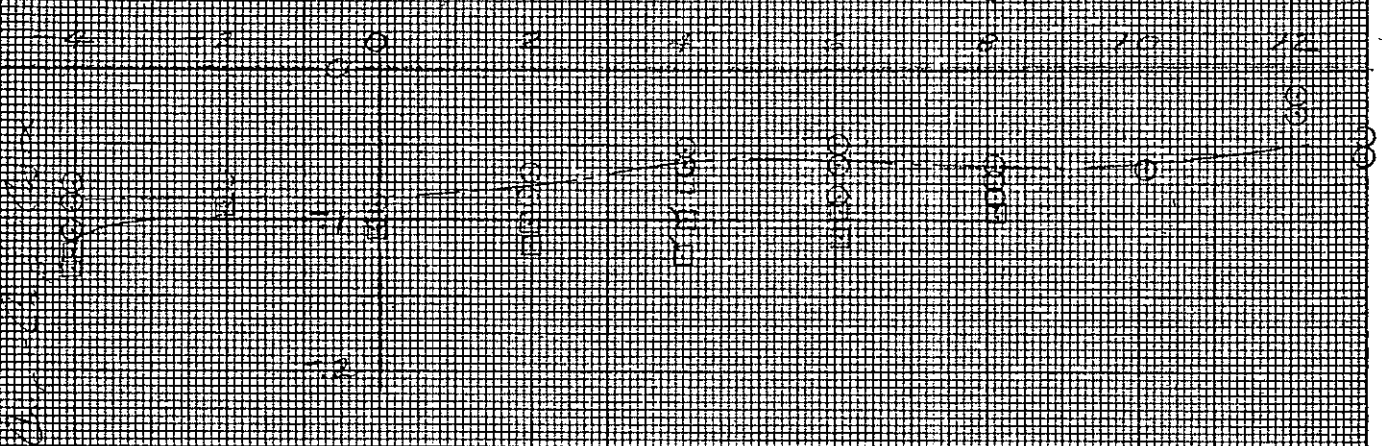
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FIG 80

ANGLE OF ATTACK α (deg)

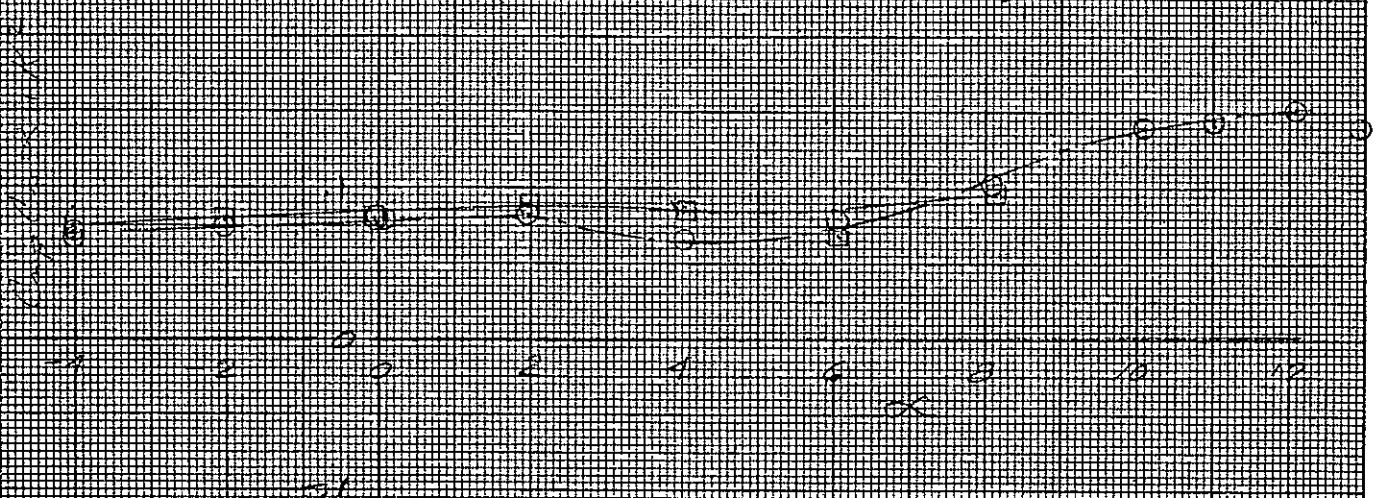
M = 55

CONFIG. EXP. RUN 1 - 54 - 4

WIND TUNNEL 0 37.25° 0 15

E 49.55° ✓

(STEEL ROD) E 55° ✓ ✓ ✓ ✓ ✓

EFFECT OF SQUARE & STEEL ROD LARS
ON LIFTING OF ROYAL DERIVATIVE

MARCH 30, 1970 CS

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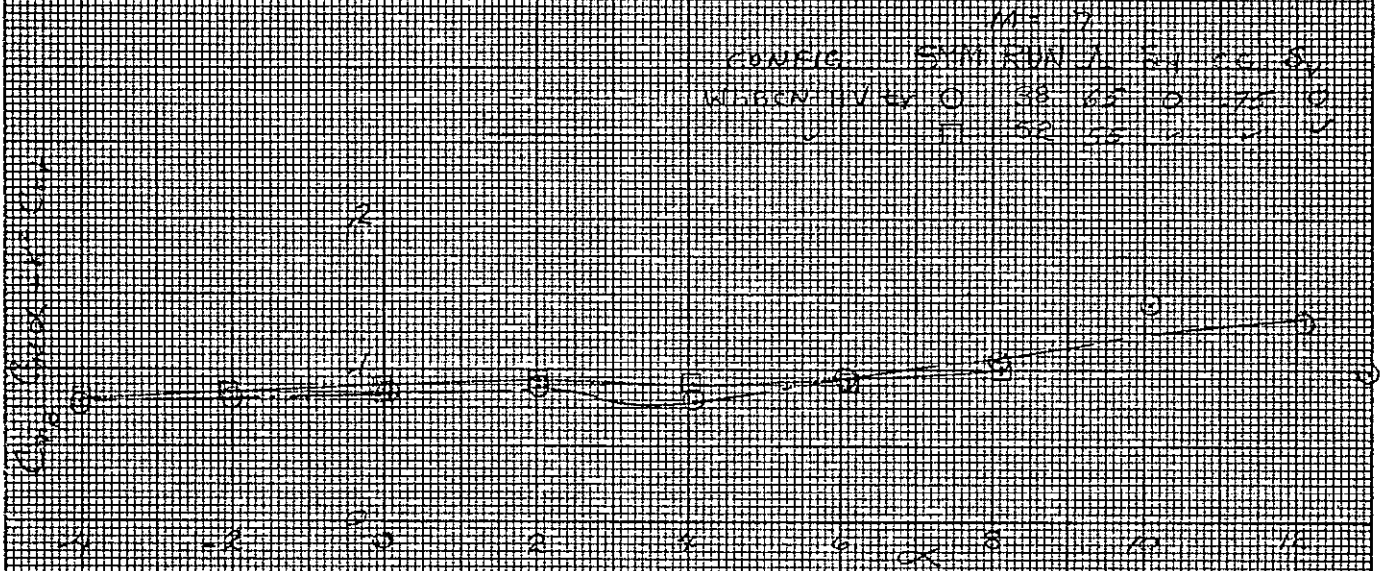
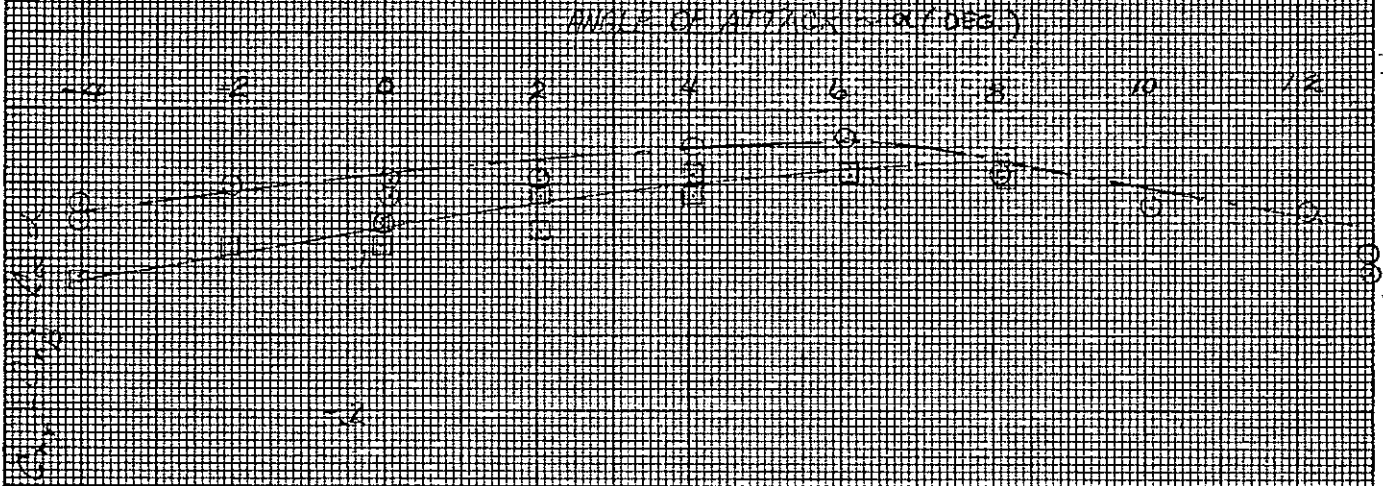
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MODEL NO

FIG. 81



EXPORT OF DATA ON GRAPHING
IN YAW DEVELOPMENT

MARCH 1967

1657 NO. 924

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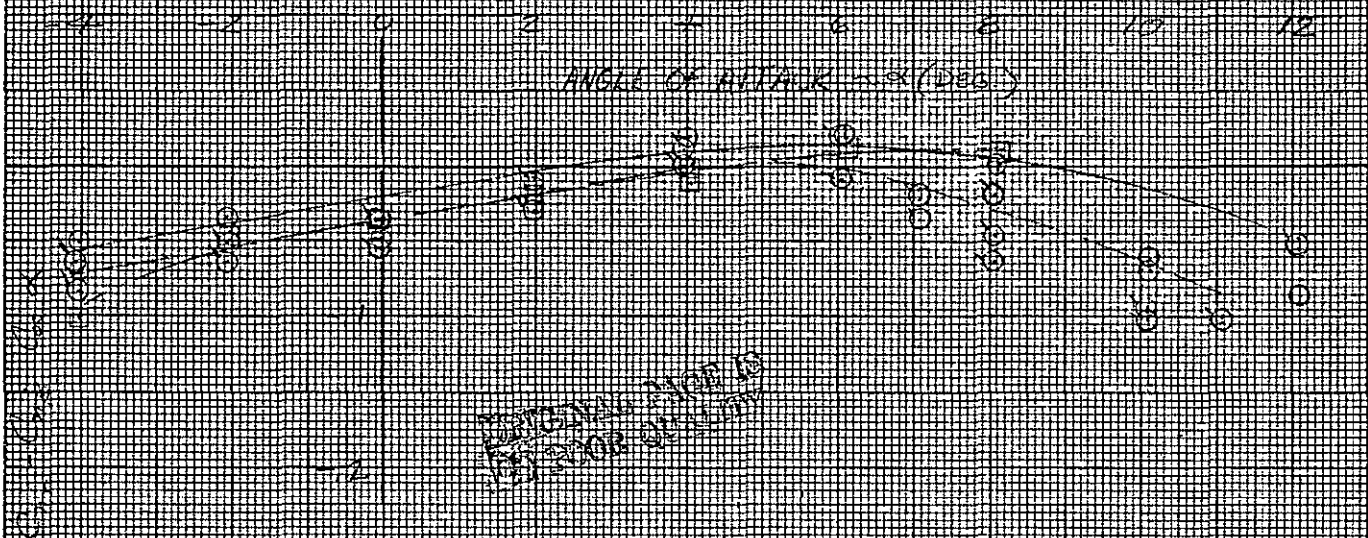
CHECKED BY:

REPORT NO NA-72-82

DATE:

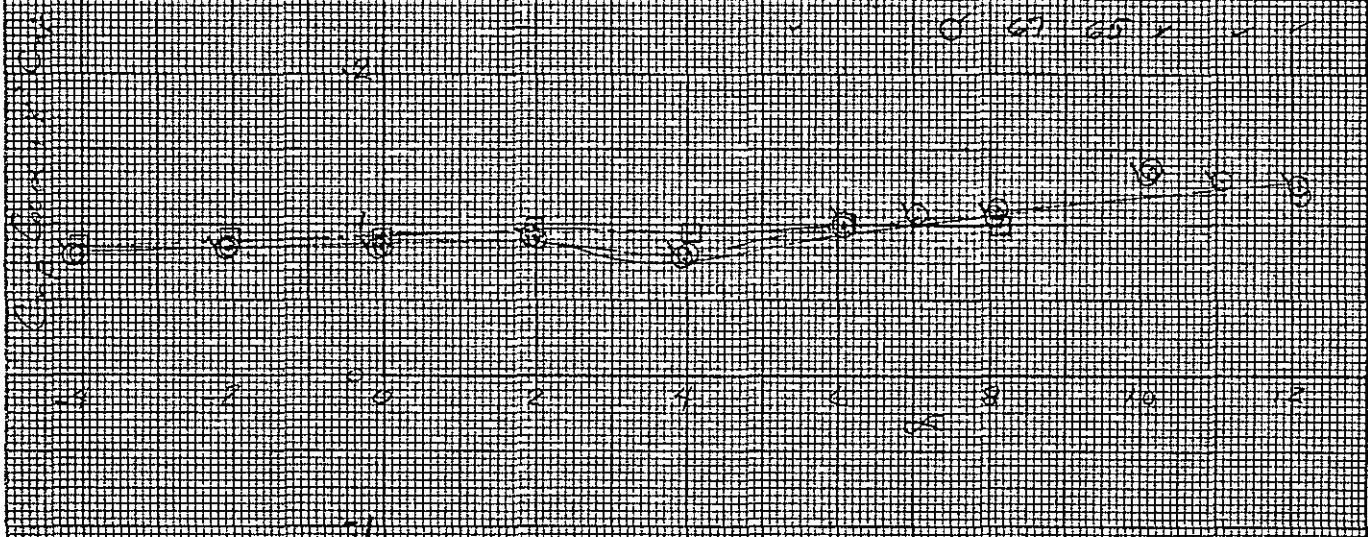
MODEL NO.

FIG 82




M = .75

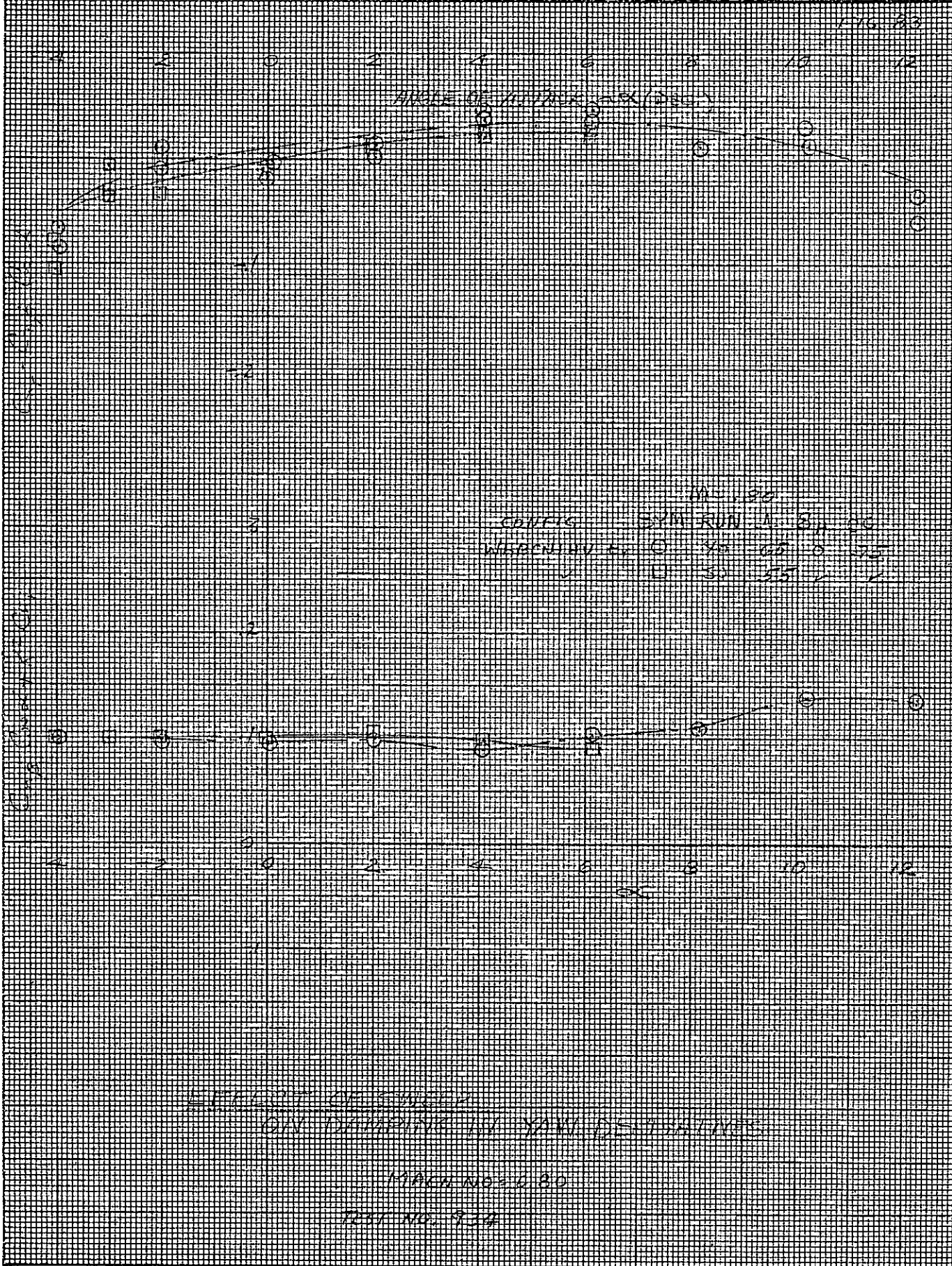
CONFIG	SYM	RVA	1	5	10	15
WING/INLET	0	39	87	0	24	0
	1	51	65	x	x	x
	0	67	65	x	x	x


EFFECT OF CHIEF
ON DAMPING IN YAW DERIVATIVES

ARCH NO - 0116

TEST NO 934

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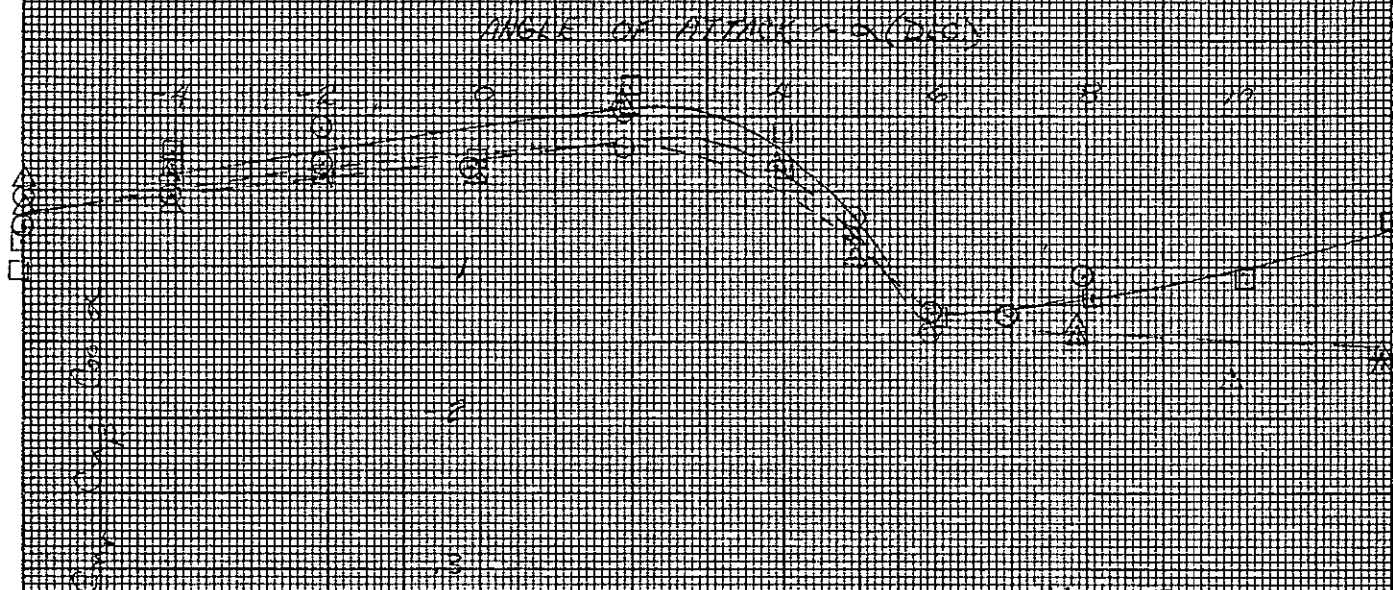
CHECKED BY:

REPORT NO. NA-72-82

DATE:

MODEL NO.

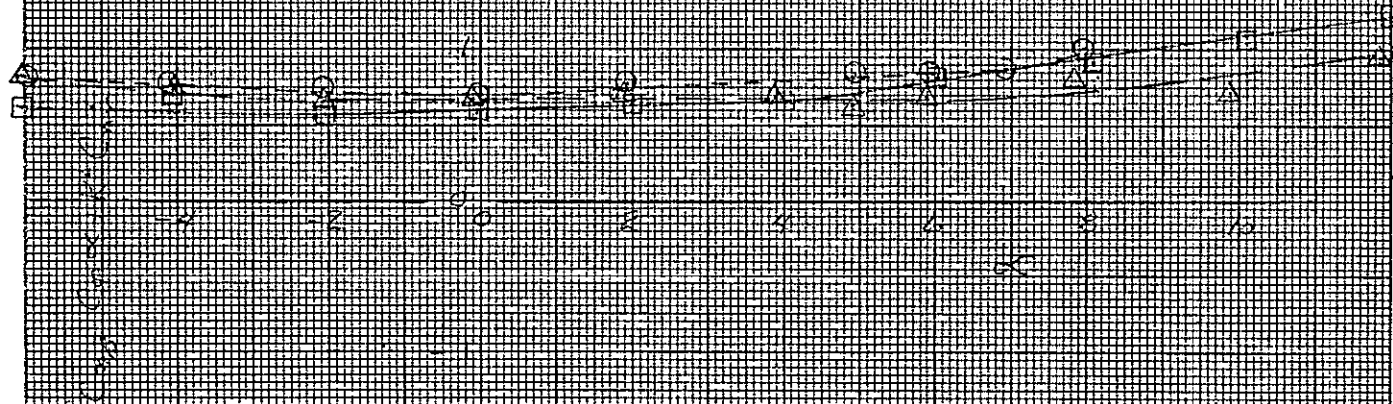
FIG. 85



W-75 CAMBER
SYN RIN SY COMP SY
 □ 20 % BCL HV □
 △ 30 % BCL HV △
 ○ 34 % BCL HV ○

NOTE

BCL DRIVE FOREBODY



COMPARISON OF LIFT

WING W-75

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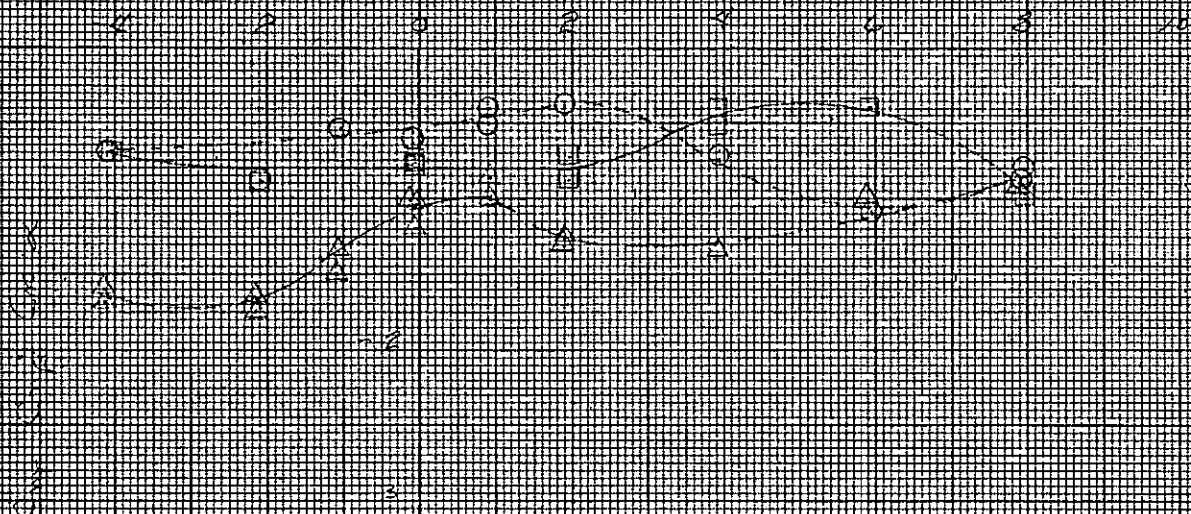
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MODEL NO.

FIG. 86

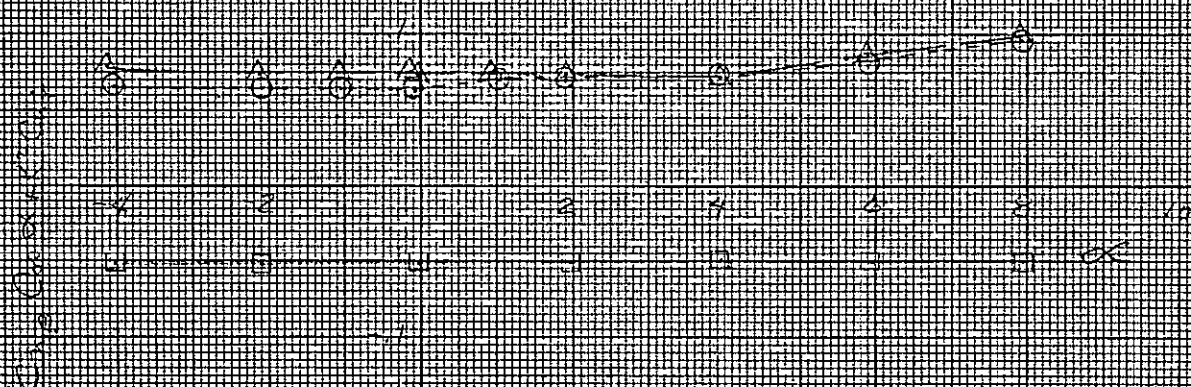
ANGLE OF ATTACK α (DEG)



N. A. R. C. - T. E.		
SYM. RUN. A. 3. COMP. 3.4		
T	30	B ₀ = 1
A	31	B ₀ = 1.0
O	32	B ₀ = 1.0

NOTE:


B₀ DRIVE FREQUENCY

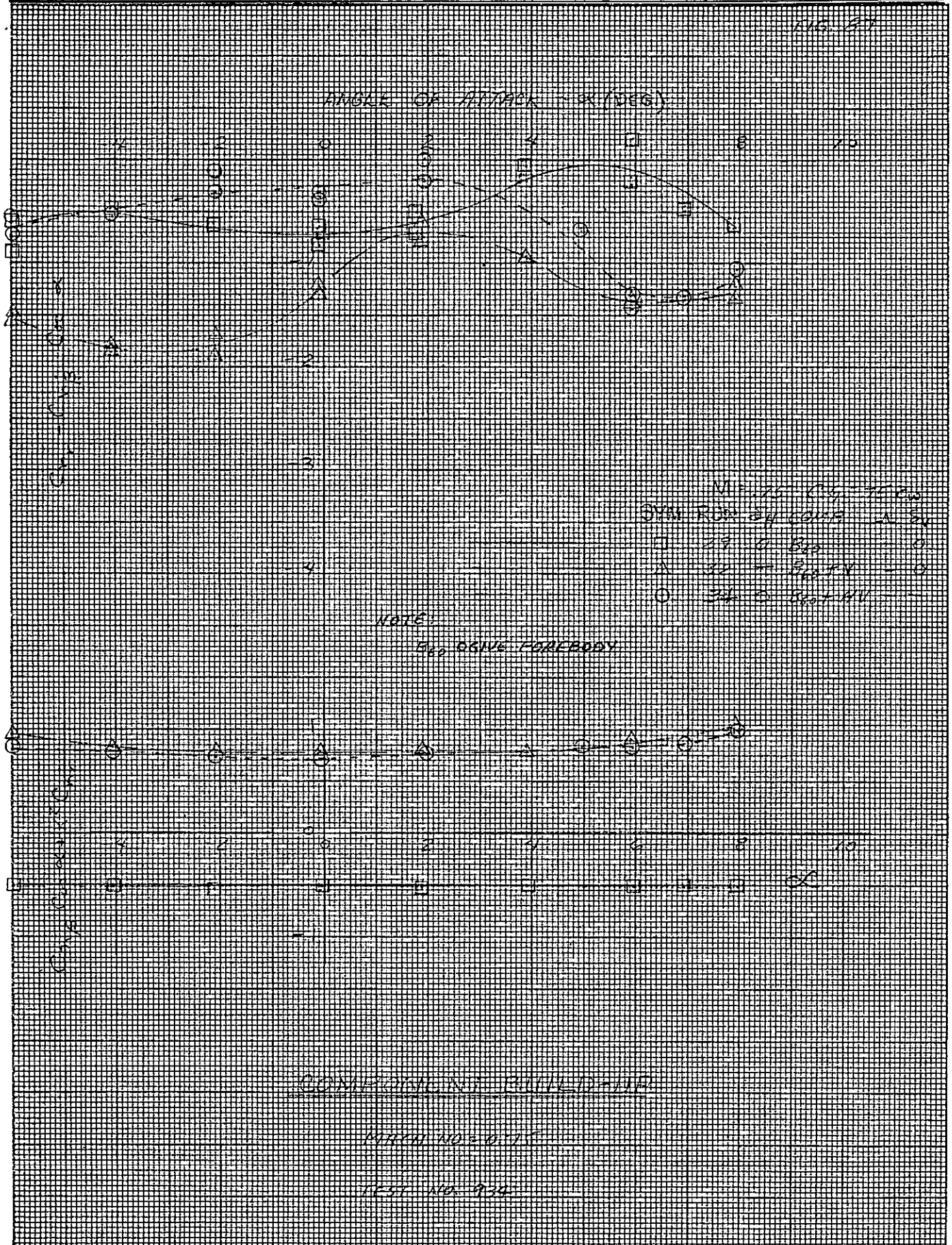


COMPONENT BUILD-UP

MODEL NO. 0.5C

REF. NO. 984

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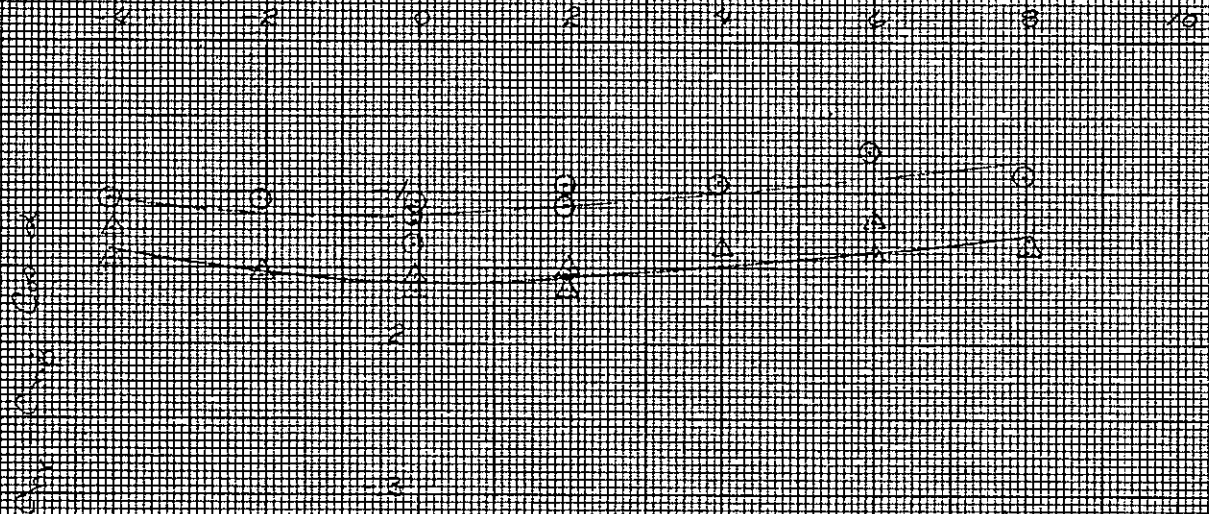
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DATE:

MODEL NO.

FIG. 53

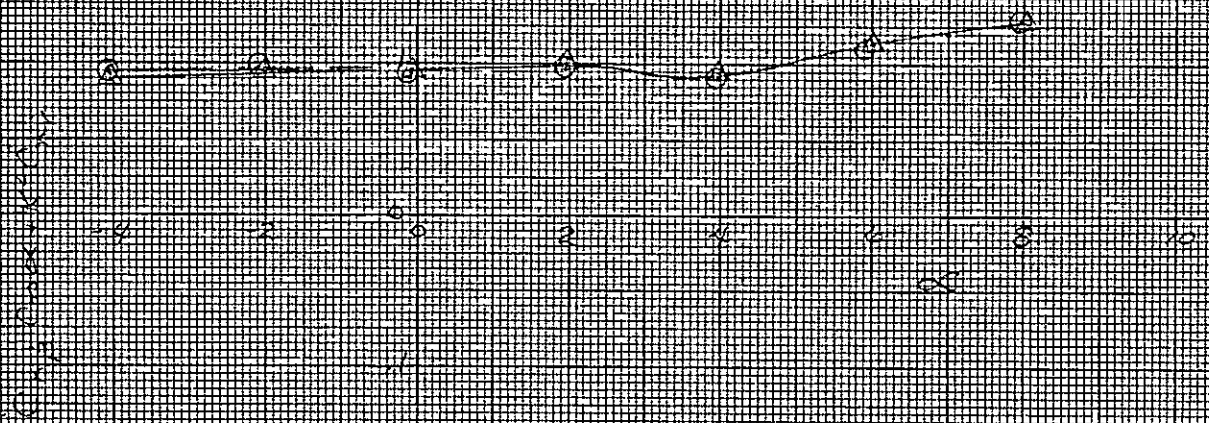
ANGLE OF ATTACK α (DEG)



ORIGINAL NOTE IS
ON COPY QUALITY

W = 1.00
CONFIG. 1 - SYM RUN 1 S. 00 84
WING ONLY W. 0 30 65 0 30 0
A 65 1 1 1 1

NOTE:
2% BODY WITH FILLER BLOCKS BETWEEN FOREWING
TAIL & FUSELAGE



DATA REPEATABILITY

W = 0.5
FIRST 1105 DASH
TEST NO 934

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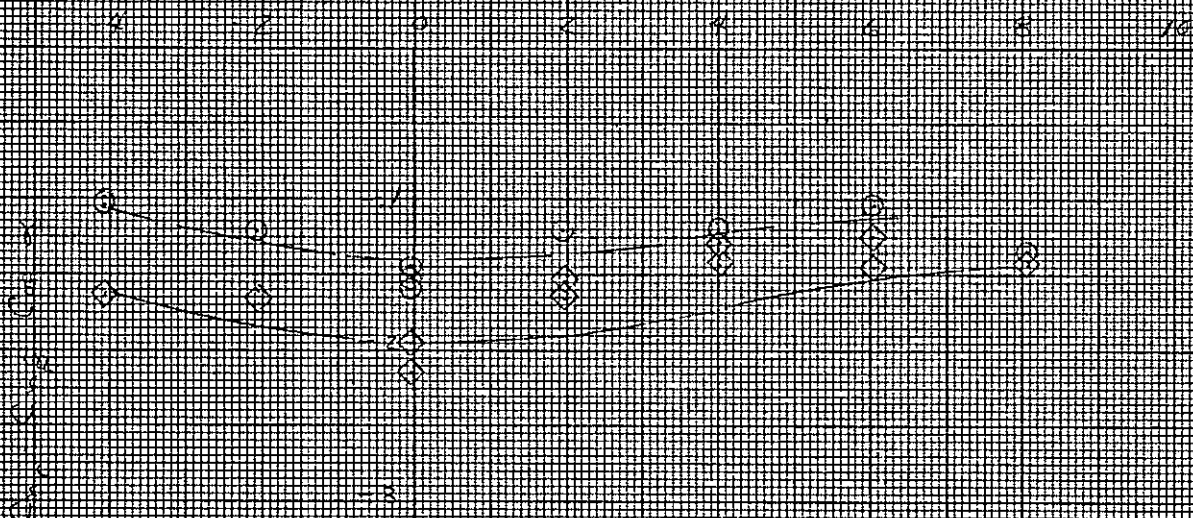
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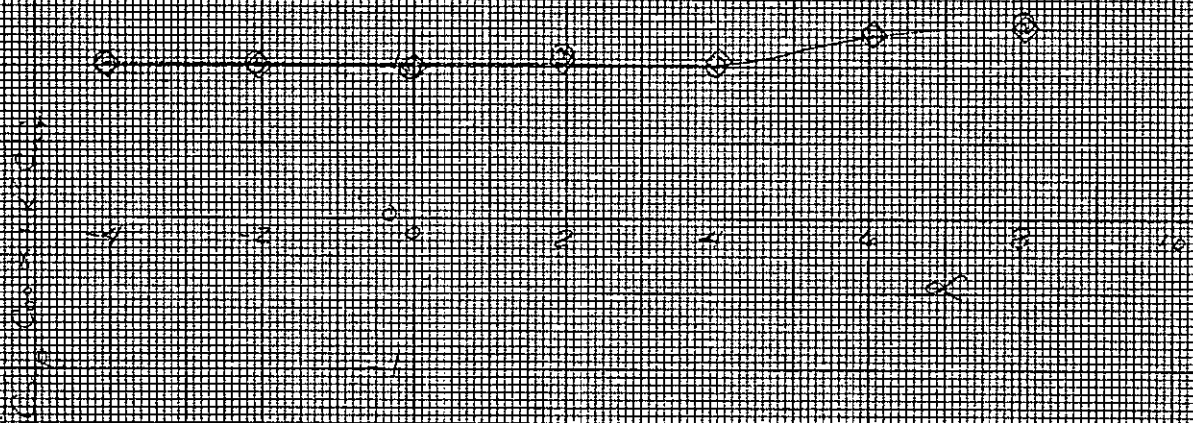
MODEL NO.

FIG 29

ANGLE OF ATTACK α (DEG)

CONF	SYM	CONF	CONF	CONF	CONF
CONF 1	CONF 2	CONF 3	CONF 4	CONF 5	CONF 6
CONF 1	CONF 2	CONF 3	CONF 4	CONF 5	CONF 6
CONF 1	CONF 2	CONF 3	CONF 4	CONF 5	CONF 6

NOTE:

BODY WITH FINE BLOCKS BEFORE RESECTION
TAIL & FUSELAGE

DATA REPEATABILITY

 $\alpha = 0.5^\circ$

MATCH NO. 0.15

TEST NO. 728


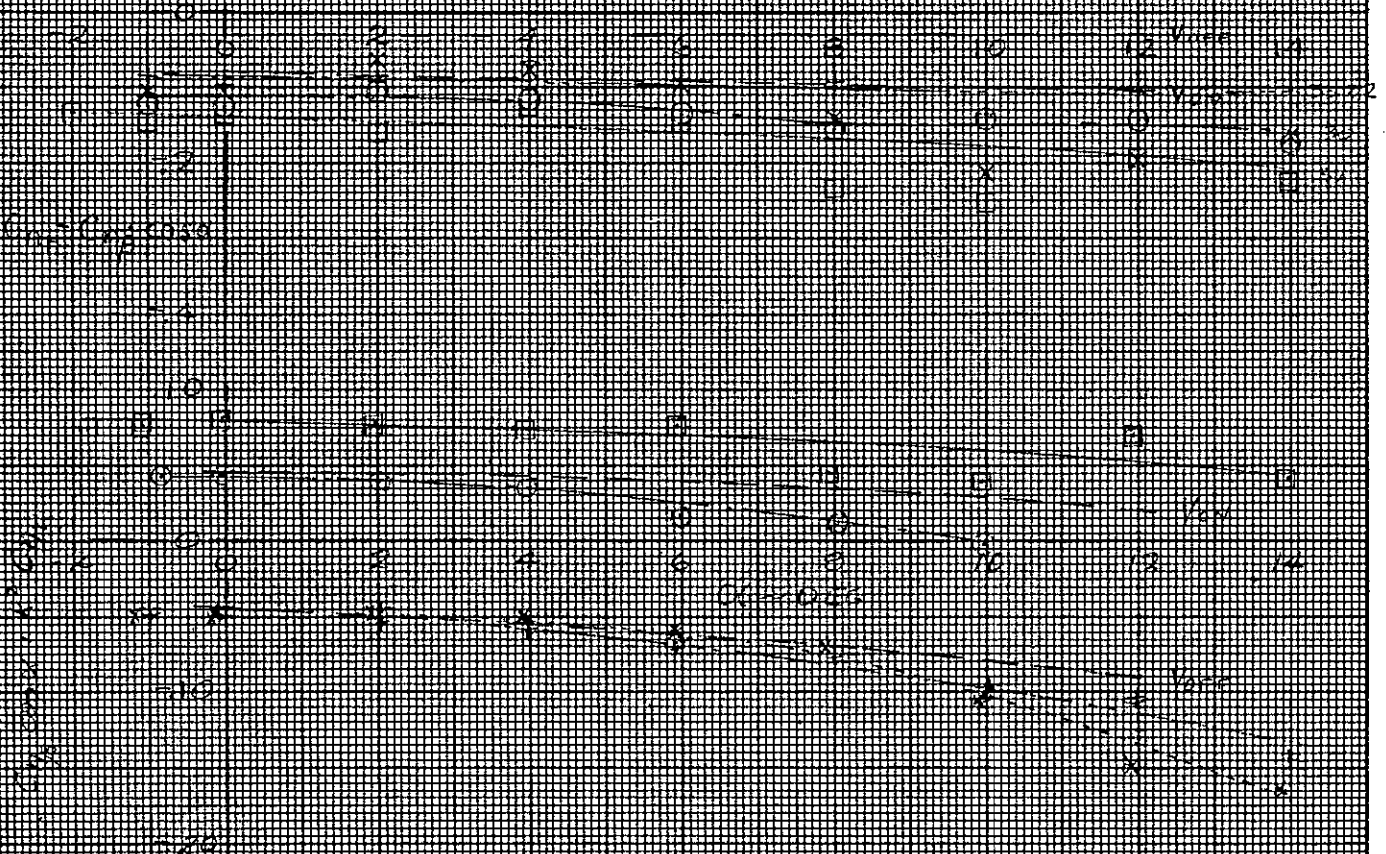
PREPARED BY:	 Los Angeles Division North American Rockwell	PAGE NO <u>116</u> OF <u>145</u>
CHECKED BY:		REPORT NO. <u>NA-72-82</u>
DATE:		MODEL NO.

FIG 72

(1) RUN OF 2000 CYCLES, AMPLITUDE 2.0G
 (2) RUN OF 2000 CYCLES, AMPLITUDE 1.0G
 (3) RUN OF 2000 CYCLES, AMPLITUDE 0.5G
 (4) RUN OF 2000 CYCLES, AMPLITUDE 0.25G
 ESTIMATED MARCH 10 1972


AREA OF A-80-4 (X) (DIRECTIONS)

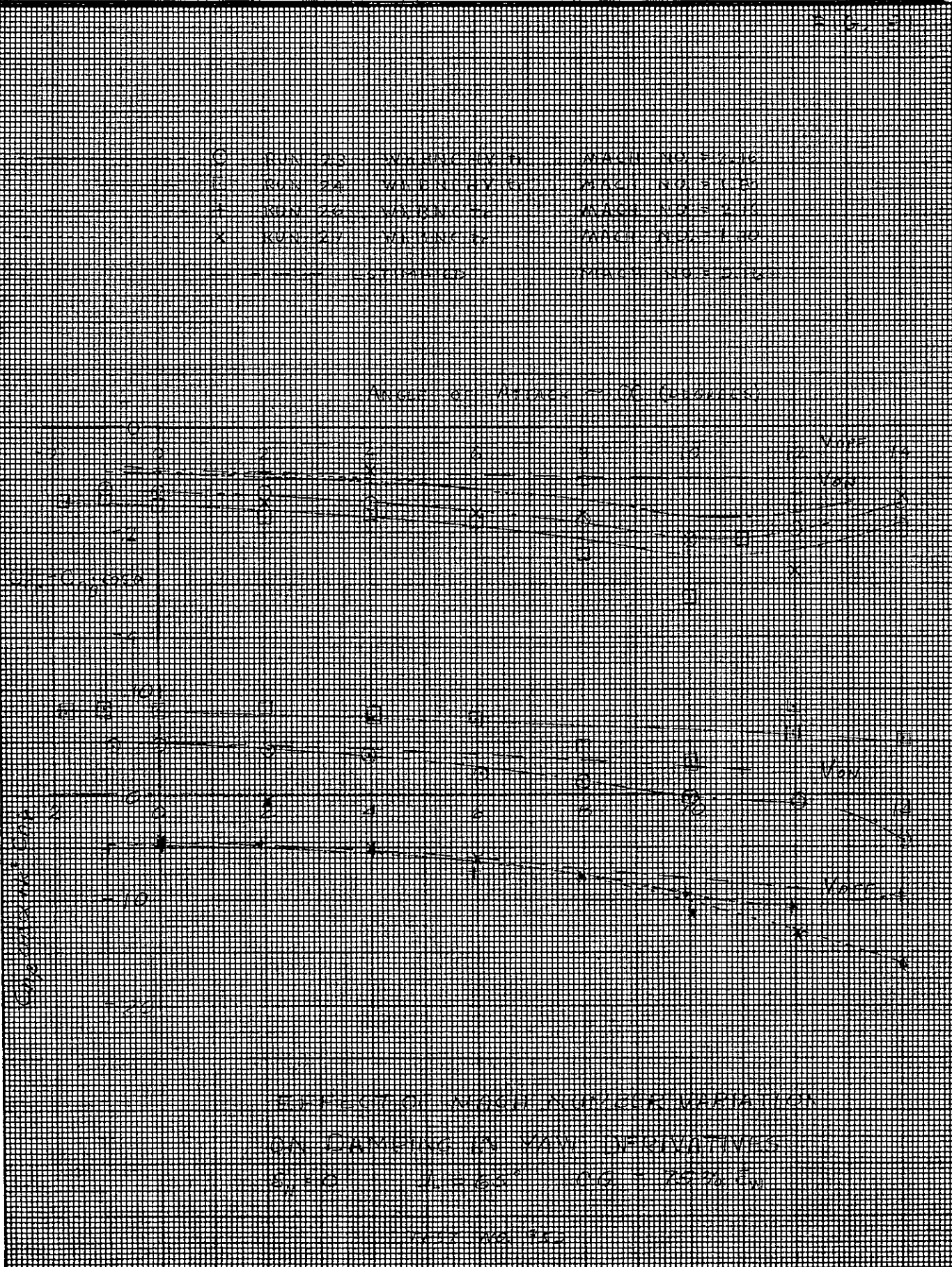


CENTER OF GRAVITY VARIATION ON
 DURING THE TEST PERIOD

AREA OF A-80-4 (X) (DIRECTIONS)

MARCH 10 1972

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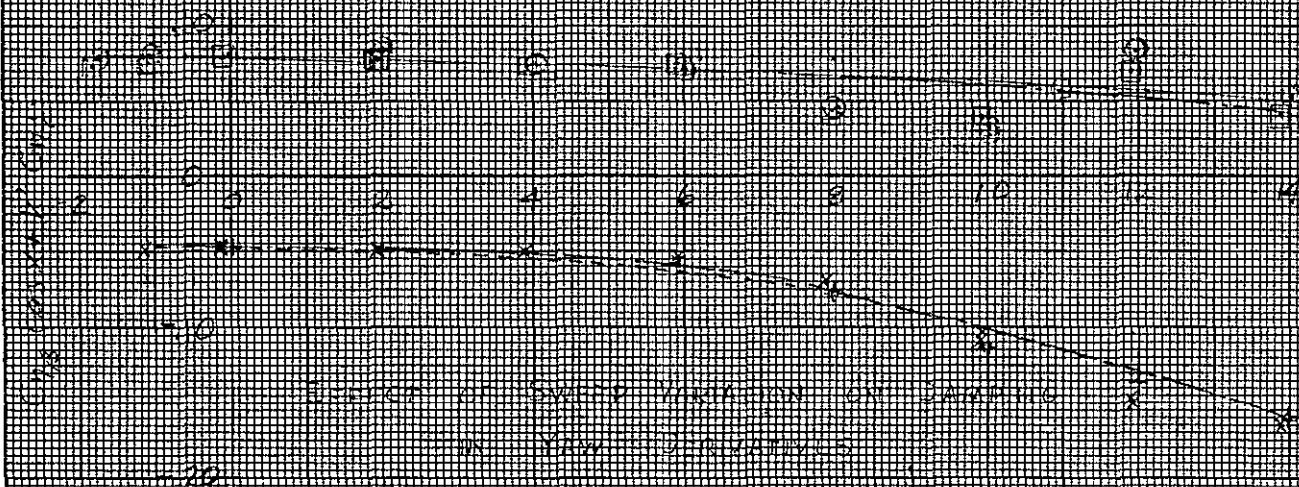
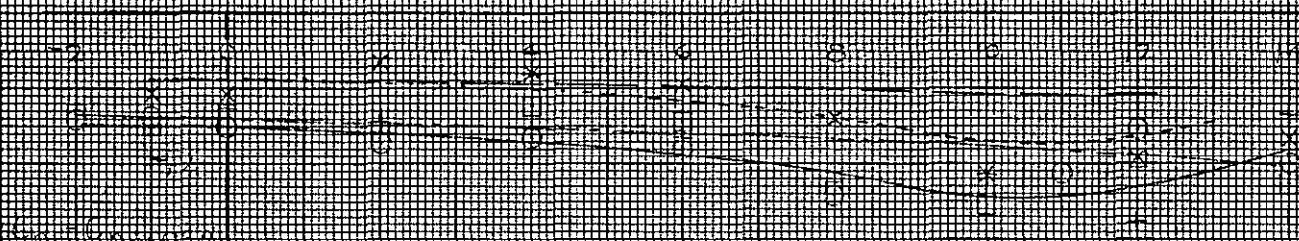
MODEL NO.

FIG 97

O RUN 24 WAGONVILLE A-565°
 T RUN 27 WAGONVILLE A-565°
 T RUN 27 WAGONVILLE A-565°
 X RUN 33 WAGONVILLE A-565°

2000
 1000
 0
 1000
 2000

ANGLE OF ATTACK (°) (GROSS)



3000
 2000
 1000
 0
 1000
 2000
 3000

PAGE NO. 118

FIG. NO. 952

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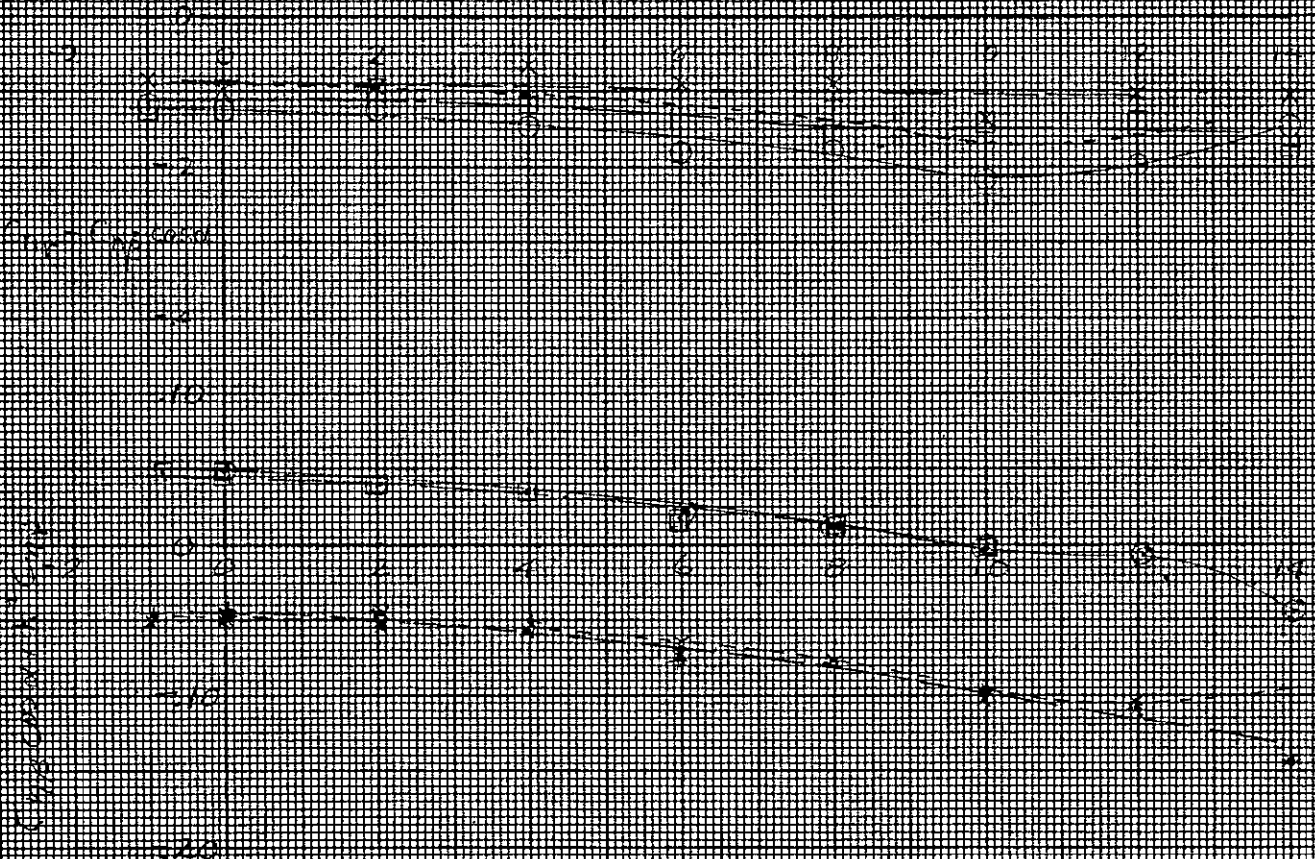
REPORT NO. NA-72-82

MODEL NO.

FIG. 93

O RUN 22 WIDENED AL-55
 * RUN 24 WIDENED AL-55
 T RUN 30 WIDENED AL-55
 X RUN 32 WIDENED AL-55

ANAL. OF A-100-00-00-00-00



EFFECT OF SURGE VARIATION ON DURATION
 OF THE JET-ENGINE

NORTH AMERICAN ROCKWELL
 LOS ANGELES, CALIF.

1001-100-00-00-00


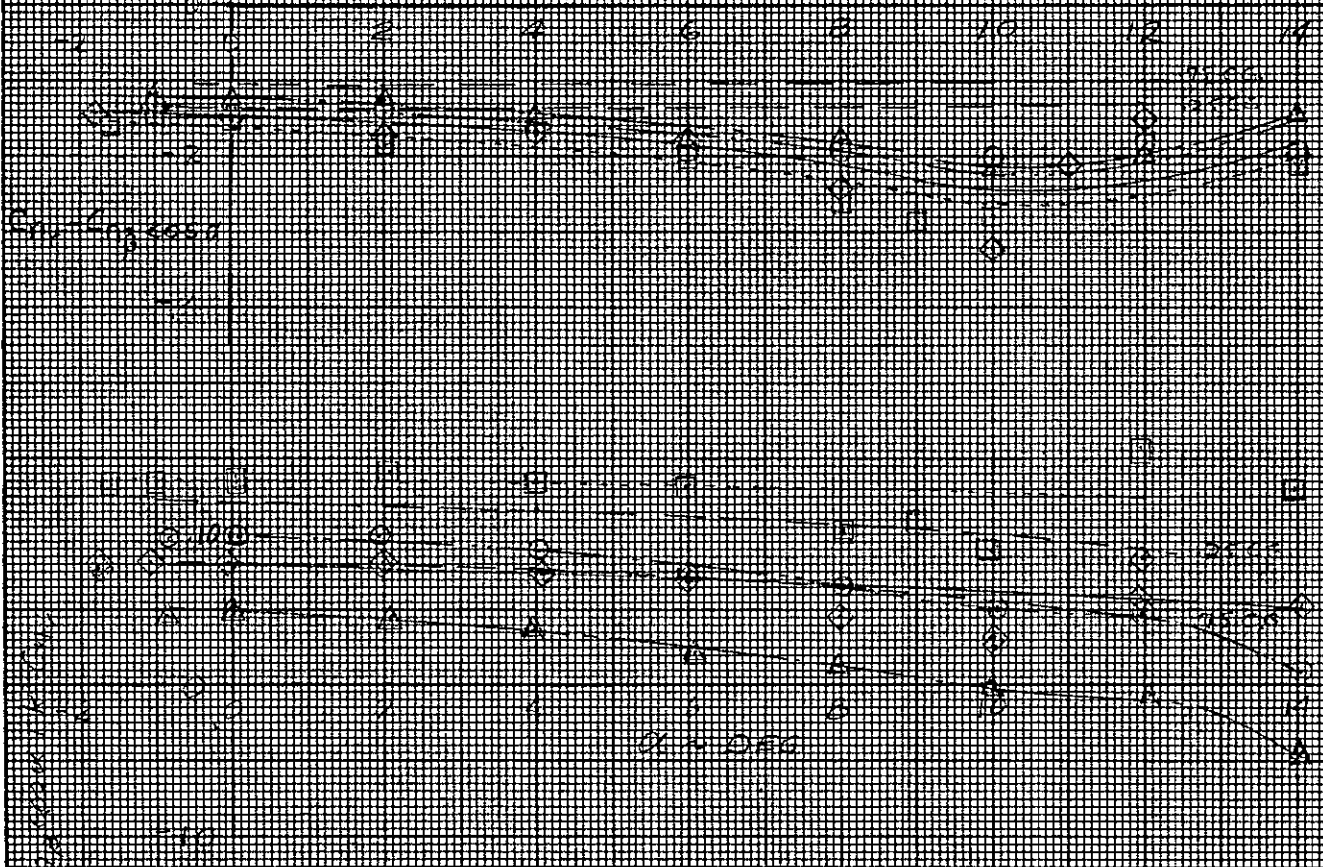
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CHECKED BY:		REPORT NO. <u>NA-72-82</u>
DATE:		MODEL NO.

FIG. 98

②	WINDING	WINDING	WINDING	WINDING
③	WINDING	WINDING	WINDING	WINDING
④	WINDING	WINDING	WINDING	WINDING
⑤	WINDING	WINDING	WINDING	WINDING

ESTIMATED TIME 10-120

ANALYSIS OF TASK - OF (10-120)



ANALYSIS OF TASK - OF (10-120)

ESTIMATED TIME 10-120

FIG. 98

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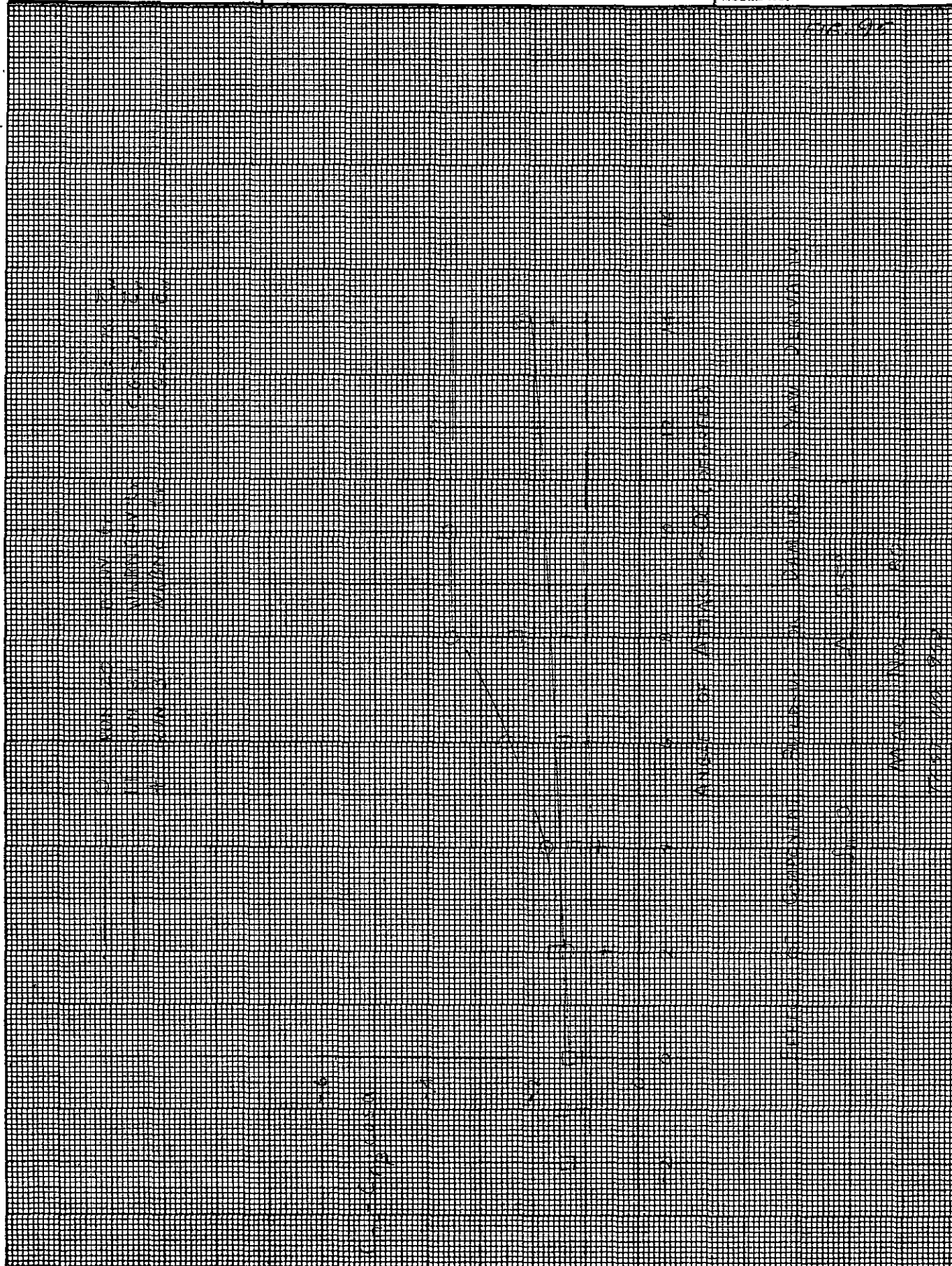
PAGE NO. 121 OF 145

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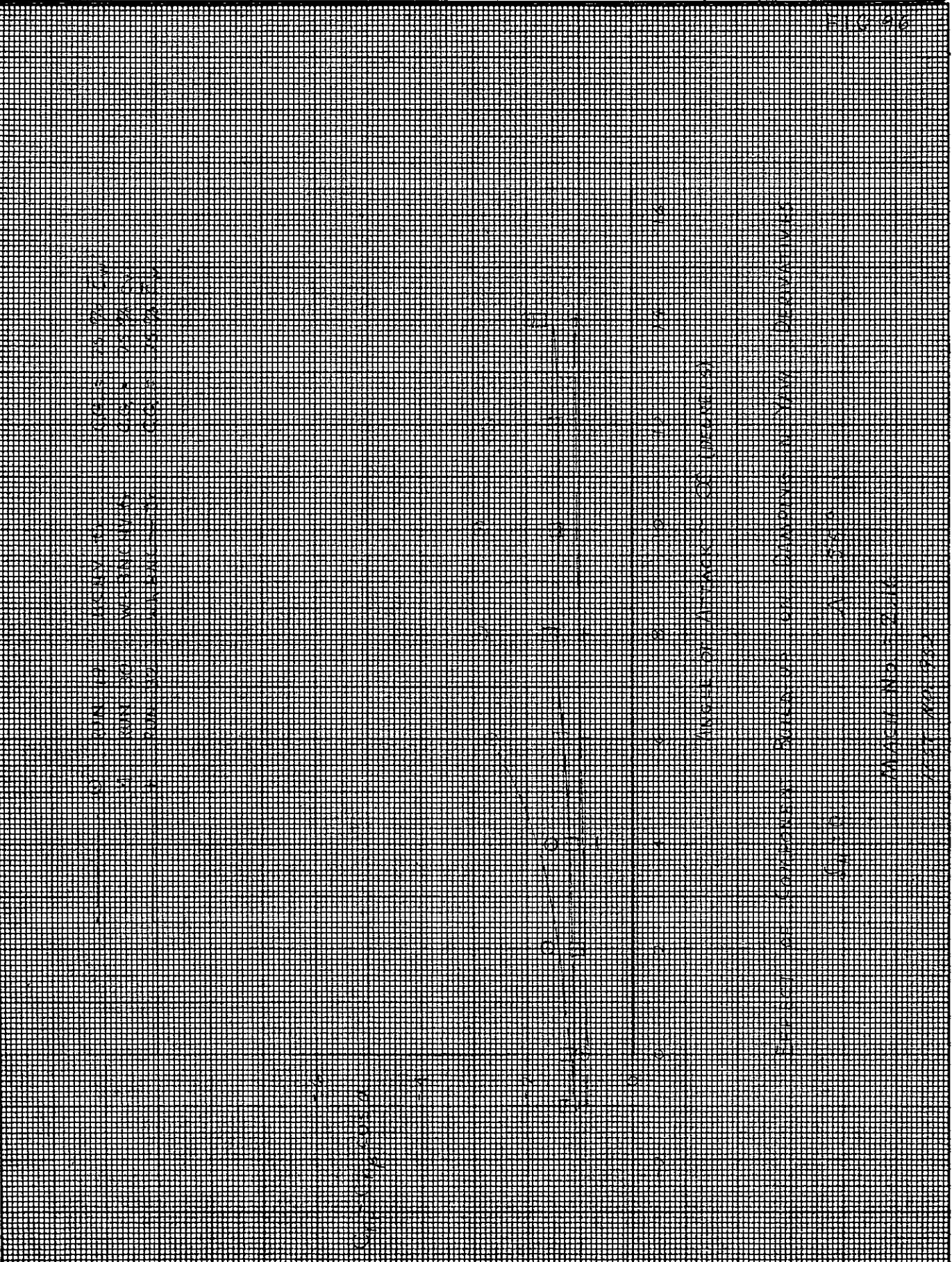


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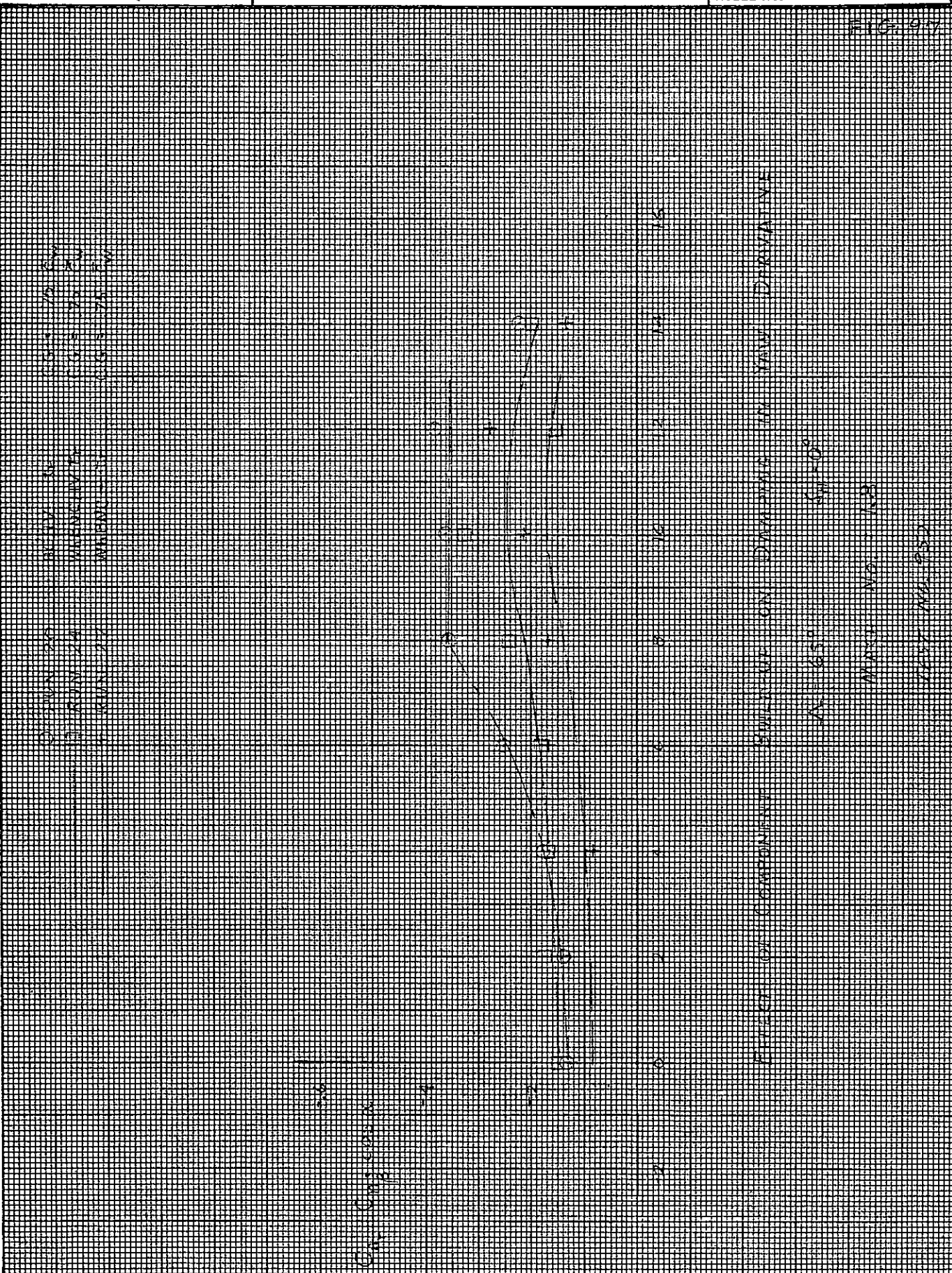
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MODEL NO.

FIG. 97



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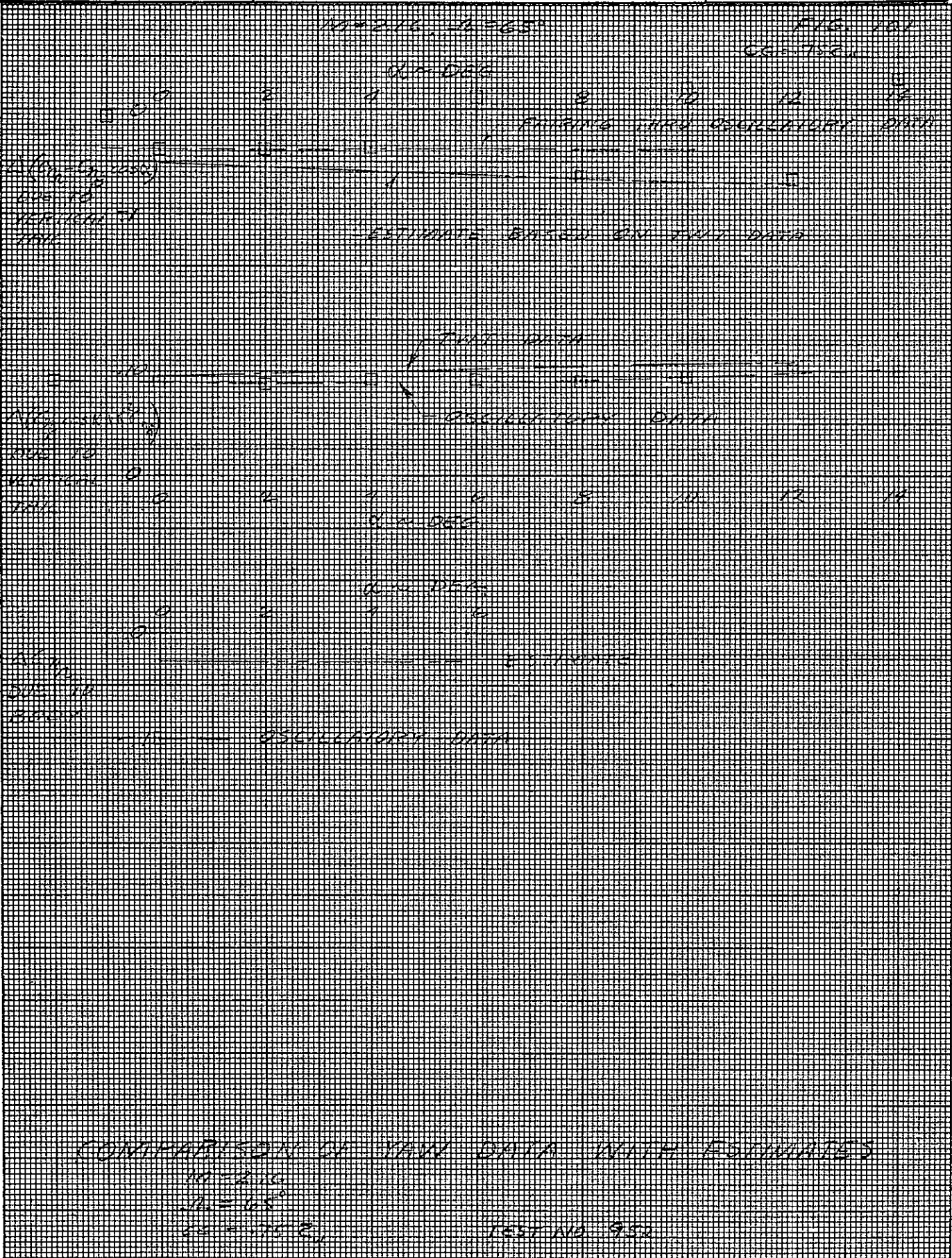
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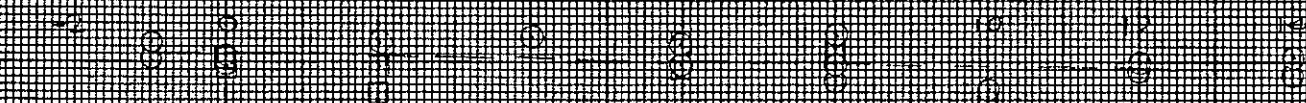
108-102

A FREQUENCY = 0 CYCLES PER SECOND

B FREQUENCY = 30 CYCLES PER SECOND

DATA 13 FOR RUN 32 (LABNO 34)

ANGLE OF ATTACK ~ 0 (DEGREES)



2
C₁ C₂ C₃ C₄ C₅ C₆ C₇ C₈ C₉ C₁₀ C₁₁ C₁₂ C₁₃ C₁₄ C₁₅ C₁₆ C₁₇ C₁₈ C₁₉ C₂₀ C₂₁ C₂₂ C₂₃ C₂₄ C₂₅ C₂₆ C₂₇ C₂₈ C₂₉ C₃₀ C₃₁ C₃₂ C₃₃ C₃₄ C₃₅ C₃₆ C₃₇ C₃₈ C₃₉ C₄₀ C₄₁ C₄₂ C₄₃ C₄₄ C₄₅ C₄₆ C₄₇ C₄₈ C₄₉ C₅₀ C₅₁ C₅₂ C₅₃ C₅₄ C₅₅ C₅₆ C₅₇ C₅₈ C₅₉ C₆₀ C₆₁ C₆₂ C₆₃ C₆₄ C₆₅ C₆₆ C₆₇ C₆₈ C₆₉ C₇₀ C₇₁ C₇₂ C₇₃ C₇₄ C₇₅ C₇₆ C₇₇ C₇₈ C₇₉ C₈₀ C₈₁ C₈₂ C₈₃ C₈₄ C₈₅ C₈₆ C₈₇ C₈₈ C₈₉ C₉₀ C₉₁ C₉₂ C₉₃ C₉₄ C₉₅ C₉₆ C₉₇ C₉₈ C₉₉ C₁₀₀ C₁₀₁ C₁₀₂ C₁₀₃ C₁₀₄ C₁₀₅ C₁₀₆ C₁₀₇ C₁₀₈ C₁₀₉ C₁₁₀ C₁₁₁ C₁₁₂ C₁₁₃ C₁₁₄ C₁₁₅ C₁₁₆ C₁₁₇ C₁₁₈ C₁₁₉ C₁₂₀ C₁₂₁ C₁₂₂ C₁₂₃ C₁₂₄ C₁₂₅ C₁₂₆ C₁₂₇ C₁₂₈ C₁₂₉ C₁₃₀ C₁₃₁ C₁₃₂ C₁₃₃ C₁₃₄ C₁₃₅ C₁₃₆ C₁₃₇ C₁₃₈ C₁₃₉ C₁₄₀ C₁₄₁ C₁₄₂ C₁₄₃ C₁₄₄ C₁₄₅ C₁₄₆ C₁₄₇ C₁₄₈ C₁₄₉ C₁₅₀ C₁₅₁ C₁₅₂ C₁₅₃ C₁₅₄ C₁₅₅ C₁₅₆ C₁₅₇ C₁₅₈ C₁₅₉ C₁₆₀ C₁₆₁ C₁₆₂ C₁₆₃ C₁₆₄ C₁₆₅ C₁₆₆ C₁₆₇ C₁₆₈ C₁₆₉ C₁₇₀ C₁₇₁ C₁₇₂ C₁₇₃ C₁₇₄ C₁₇₅ C₁₇₆ C₁₇₇ C₁₇₈ C₁₇₉ C₁₈₀ C₁₈₁ C₁₈₂ C₁₈₃ C₁₈₄ C₁₈₅ C₁₈₆ C₁₈₇ C₁₈₈ C₁₈₉ C₁₉₀ C₁₉₁ C₁₉₂ C₁₉₃ C₁₉₄ C₁₉₅ C₁₉₆ C₁₉₇ C₁₉₈ C₁₉₉ C₂₀₀ C₂₀₁ C₂₀₂ C₂₀₃ C₂₀₄ C₂₀₅ C₂₀₆ C₂₀₇ C₂₀₈ C₂₀₉ C₂₁₀ C₂₁₁ C₂₁₂ C₂₁₃ C₂₁₄ C₂₁₅ C₂₁₆ C₂₁₇ C₂₁₈ C₂₁₉ C₂₂₀ C₂₂₁ 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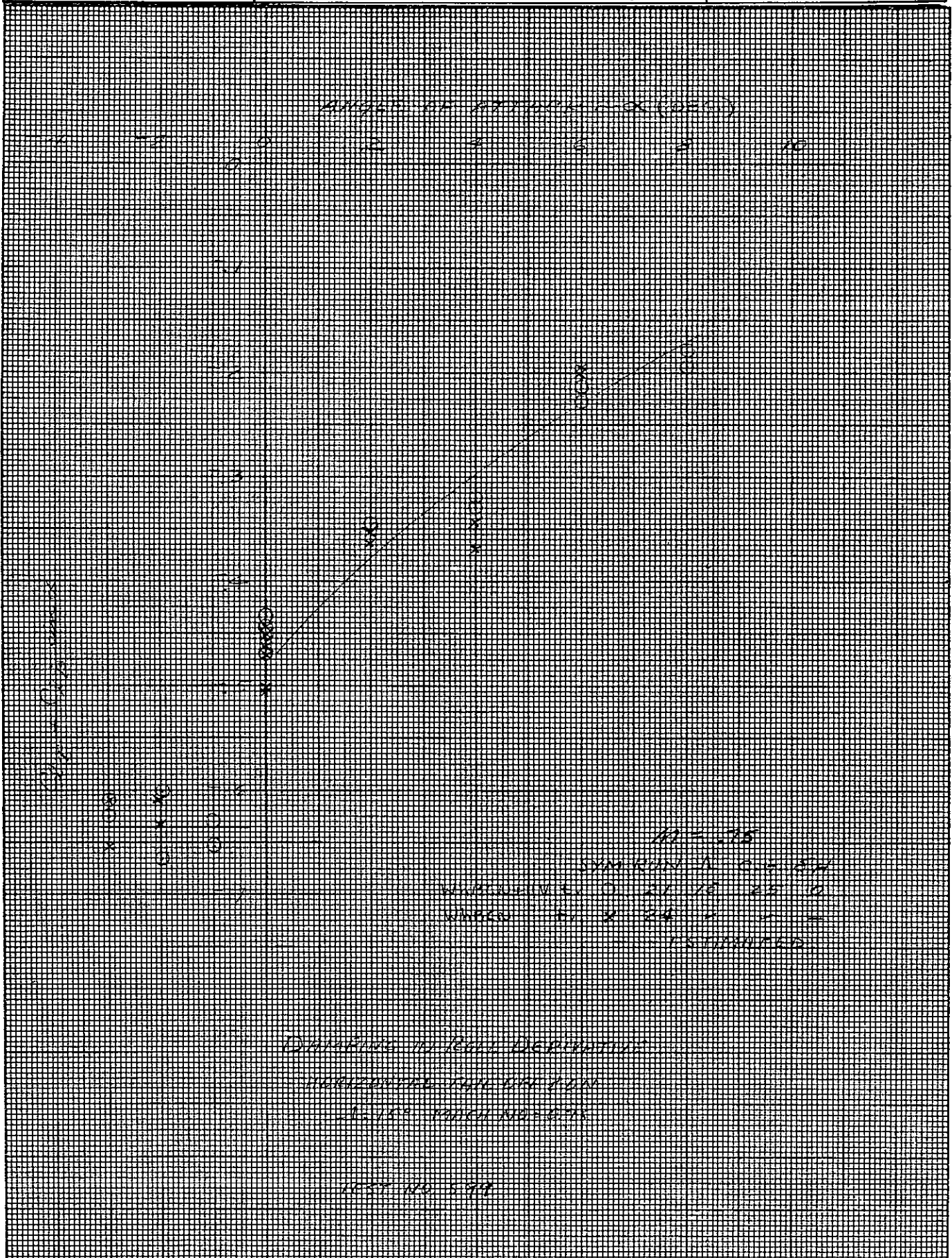
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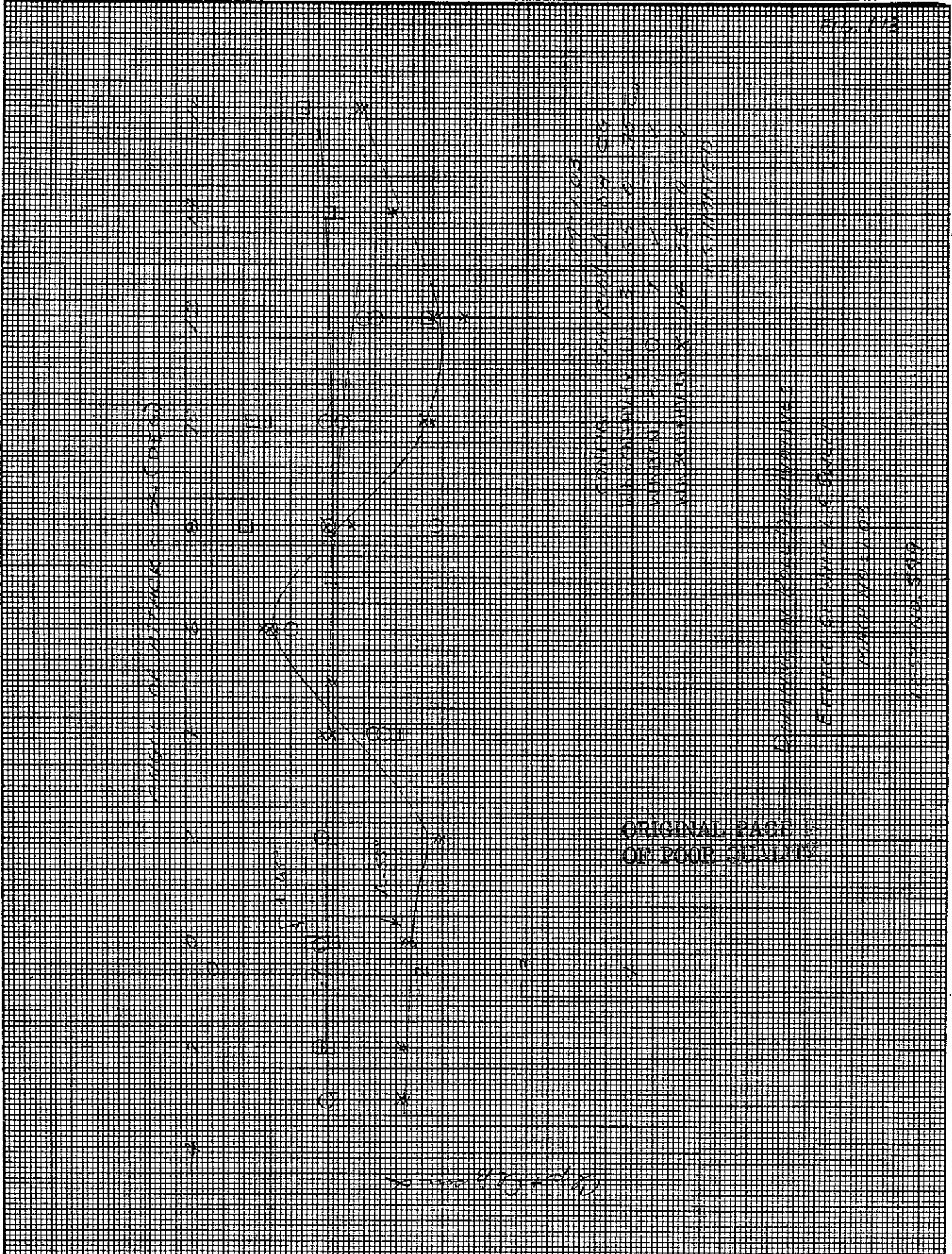


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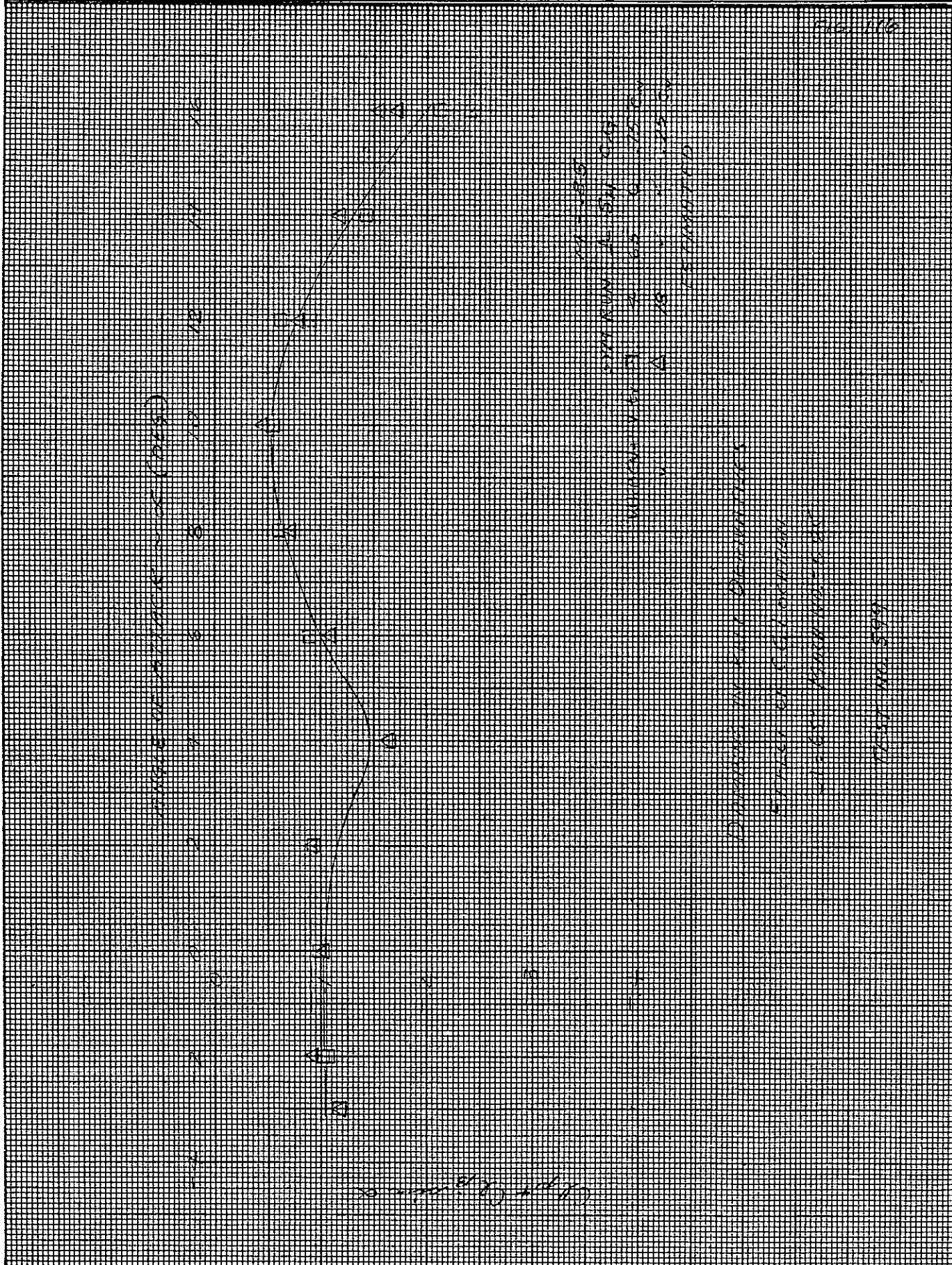


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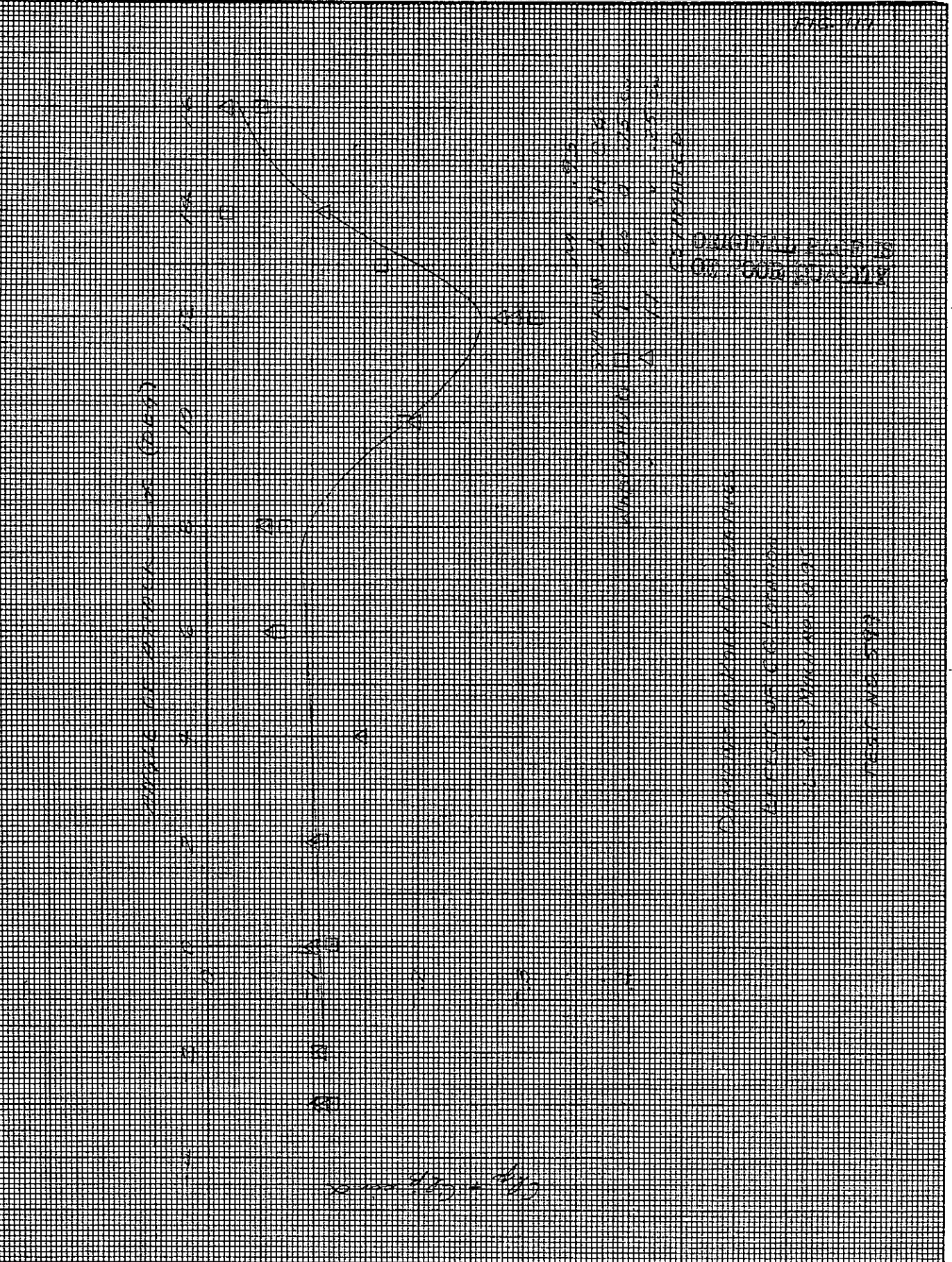


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